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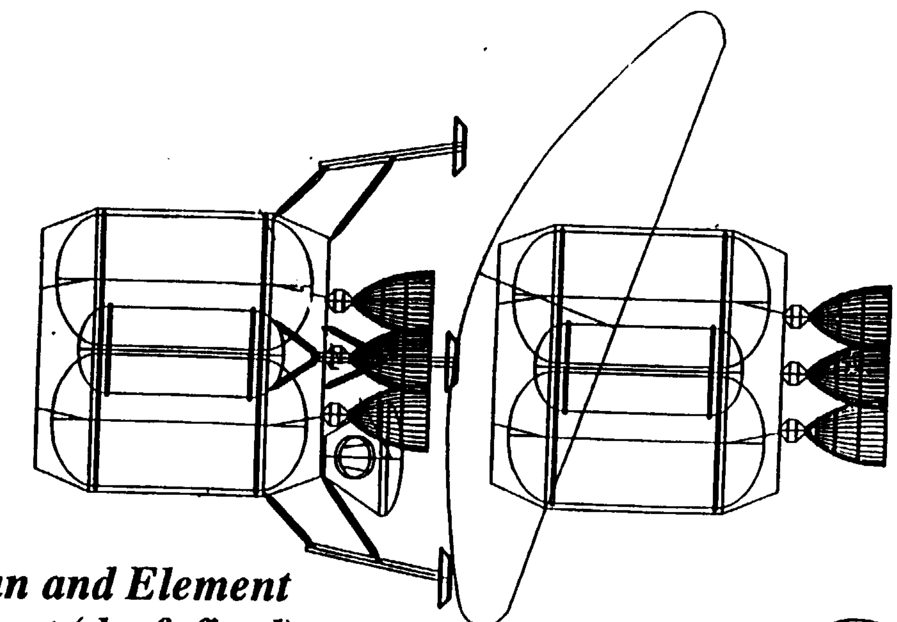
Space Transfer Concepts and Analysis for Exploration Missions



Implementation Plan and Element Description Document (draft final)
Volume 6: Lunar Systems

March 8, 1991

ADVANCED CIVIL SPACE SYSTEMS

Boeing Aerospace and Electronics
Huntsville, Alabama

NASA Contract NAS8-37857

D615-10026-6

Space Transfer Concepts and Analyses for Exploration Missions

NASA Contract NAS8-37857

Lunar Transportation Family
Implementation Plan and Element Description Document

Boeing Aerospace and Electronics
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Documentation Set:
D615-10026-1 IP and ED Volume 1: Major Trades, Books 1 and 2
D615-10026-2 IP and ED Volume 2: Cryogenic/Aerobrake Vehicle
D615-10026-3 IP and ED Volume 3: Nuclear Thermal Rocket Vehicle
D615-10026-4 IP and ED Volume 4: Solar Electric Propulsion Vehicle
D615-10026-5 IP and ED Volume 5: Nuclear Electric Propulsion Vehicle
D615-10026-6 IP and ED Volume 6: Lunar Transportation Family

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Lunar Transportation Family
Implementation Plan and Element Description Document

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Symbols, Abbreviations and Acronyms

ACRV	Advanced crew recovery vehicle
ACS	Attitude control system
AFE	Aerobrake Flight Experiment
A&I	Attachment and integration
Al	Aluminum
ALARA	As low as reasonably achievable
ALS	Advanced Launch System
ALSPE	Anomalously large solar proton event
am	Atomic mass (unit)
AR	Area ratio
ARGPER	Argument of perigee
ARS	Atmospheric revitalization system
art-g	Artificial gravity
asc	Ascent
ASE	Advanced space engine
AU	Astronomical Unit (=149.6 million km)
BIT	Built-in test
BITE	Built-in test equipment
BLAP	Boundary Layer Analysis Program
BFO	Blood-forming organs
BMR	Body mounted radiator
C	Degrees Celsius
CAB	Cryogenic/aerobrake
CAD/CAM	Compter-aided design/computer-aided manufacturing
CAP	Cryogenic all-propulsive
C_d	Drag coefficient
CELSS	Closed Environmental Life Support System
CHC	Crew health care
CG	Center of gravity
C_L	Lift coefficient
cm	Centimeter = 0.01 meter
c/m	Crew module
CM	Center of mass
c/o	Check out
C of F	Cost of facilities
conj	Conjunction
COSPAR	Committee on Space Research of the International Council of Scientific Unions
CO2	Carbon dioxide
Cryo	Cryogenic
C3	Hyperbolic excess velocity squared (in km^2/s^2)
C&T	Communications and Telemetry
CTV	Cargo Transport Vehicle (operates in Earth orbit)
d	days
DDT&E	Design, development, testing, and evaluation
DE	Dose equivalent
deg	Degrees
desc	Descent

DMS	Data management system
dV	Velocity change (ΔV)
EA	Earth arrival
E arr	Earth arrival
Ec	Modulus of elasticity in compression
ECCV	Earth crew capture vehicle
ECWS	Element control work station
ECLSS	Environment control and life support system
EP	Electric propulsion
ESA	European Space Agency
e.s.o.	Engine start opportunity
ET	External Tank
ETO	Earth-to-orbit
EVA	Extra-vehicular activity
F_c	Circulation efficiency factor
FD&D	Fire Detection and Differentiation
F_{ew}	Life support weight factor
FEL	First element launch
F_f	Specific floor count factor
F_{fa}	Specific floor area factor
F_i	Aerobrake integration factor
F_l	Specific length factor
F_n	Normalized spatial unit count factor
F_o	Path options factor
F_p	Useful perimeter factor
F_{pc}	Parts count factor
F_{pr}	Proximity convenience factor
F_{rp}	Plan aspect ratio factor
F_{rs}	Section aspect ratio factor
FSE	Flight support equipment
F_s	Vault factor
F_{ss}	Safe-haven split factor
F_u	Spatial unit number factor
F_v	Volume range factor
FY88	Fiscal Year 1988 (=October 1, 1987 to September 30, 1988. Similarly for other years)
g	Acceleration in Earth gravities (=acceleration/9.80665m/s^2)
GCNR	Gas core nuclear rocket
GCR	Galactic cosmic rays
GEO	Geosynchronous Earth Orbit
GN2	Gaseous nitrogen
GN&C	Guidance, navigation, and control
GPS	Global Positioning System
Gy	Gray (SI unit of absorbed radiation energy = 10^4 erg/gm)
hab	Habitation
HD	High Density
HEI	Human Exploration Initiative (obsolete for SEI)
HLLV	Heavy lift launch vehicle
hrs	Hours

hyg w	Hygeine water
HZE	High atomic number and energy particle
H2	Hydrogen
H_2O	Water
ICRP	International Commission on Radiation Protection
IMLEO	Initial mass in low Earth orbit
in.	Inches
inb	Inbound
IP&ED	Implementation Plan and Element Description
IR&D	Independant research and development
Isp	Specific impulse (=thrust/mass flow rate)
ISRU	In-situ resource utilization
JEM	Japan Experiment Module (of SSF)
JSC	Johnson Space Center
k	klb
keV	Thousand electron volt
kg	Kilograms
klb	Kilopounds (thousands of pounds. Conversion to SI units=4448 N/klb)
klbf	Kilopound force
km	Kilometers
KM	Kilometers
KM/Sec	Kilometers per second
KM/SEC	Kilometers per second
ksi	Kilopounds per square inch
LCC	Life cycle cost
L/D	Lift-to-drag ratio
LD	Low density
LDM	Long duration mission
LEO	Low Earth orbit
LET	Linear energy transfer
LEV	Lunar excursion vehicle
LEVCM	Lunar excursion vehicle crew module
Level II	Space Exploration Initiative project office, Johnson Space Center
LH2	Liquid hydrogen
LiOH	Lithium hydroxide
LLO	Low Lunar orbit
LM	Lunar Module
LOR	Lunar orbit rendezvous
LOX	Liquid oxygen
LS	Lunar surface
LTV	Lunar transfer vehicle
LTVCM	Lunar transfer vehicle crew module
L2	Lagrange point 2. A point behind the Moon as seen from the Earth which has the same orbital period as the moon.
m	Meters
[MarsGram	Western Union interplanetary telegram]
[MARSIN	Martian pornography]
MASE	Mission analysis and systems engineering (same as Level II q.v.)
MAV	Mars ascent vehicle

M/C_DA	Ballistic coefficient (mass / drag coefficient times area)
MCRV	Modified crew recovery vehicle
m_e	Mass of electron
MEOP	Maximum expected operating pressure
MeV	Million electron volt
MEV	Mars excursion vehicle
MLI	Multi-layer insulation
mm	Millimeter (=0.001 meter)
MMH	Monomethylhydrazine
MMV	Manned Mars vehicle
MOC	Mars orbit capture
MOI	Mars orbit insertion
mod	Module
M&P	Materials and processes
MPS	Main propulsion system
MR	Mixture ratio
m/sec	Meters per second
MSFC	Marshall Space Flight Center
Msi	Million pounds per square inch
mt	Metric tons (thousands of kilograms)
mT	Metric tons
MTBF	Mean time between failures
MTV	Mars transfer vehicle
MWe	Megawatts electric
m^3	Cubic Meters
N	Newton. Kilogram-meters per second squared
n/a	Not applicable
NASA	National Aeronautics and Space Administration
NCRP	National Council on Radiation Protection
NEP	Nuclear-electric propulsion
NERVA	Nuclear engine for rocket vehicle application
NTP	Nuclear thermal propulsion (same as NTR)
NSO	Nuclear safe orbit
NTR	Nuclear thermal rocket
N2O4	Nitrogen tetroxide
OSE	Orbital support equipment
OTIS	Optimal Trajectories by Implicit Simulation program
outb	Outbound
O2	Oxygen
PBR	Particle bed reactor
Pc	Chamber pressure
PEEK	Polyether-ether ketone
PEGA	Powered Earth gravity assist
P/L	Payload
POTV	Personnel orbital transfer vehicle
pot w	Potable water
PPU	Power processing unit
prop	Propellant
psi	Pounds per square inch
PV	Photovoltaic

Q	Heat flux (Joules per square centimeter)
Q	Radiation quality factor
RAAN	Right ascension of ascending node
RCS	Reaction control system
Re	Reynolds number
RF	Radio frequency
RMLEO	Resupply mass in low Earth orbit
ROI	Return on investment
RPM	Revolutions per minute
RWA	Relative wind angle
R&D	Research and Development
	Rendezvous and dock
SAA	South Atlantic Anomaly
SAIC	Science Applications International Corporation
SEI	Space Exploration Initiative
SEP	Solar-electric propulsion
SI	International system of units (metric system)
SiC	Silicon carbide
SMA	Semimajor axis
sol	Solar day (24.6 hours for Mars)
SPE	Soalr proton events
SRB	Solid Rocket Booster
SSF	Space Station Freedom
SSME	Space Shuttle Main Engine
STCAEM	Space Transfer Concepts and Analysis for Exploration Missions
stg	Stage
surf	Surface
Sv	Sieviert (SI unit of dose equivalent = Gy x Q)
S1	Distance along aerobrake surface forward of the stagnation point
S2	Distance along aerobrake surface aft of the stagnation point
S3	Distance along aerobrake surface starboard of the stagnation point
t.	Metric tons (1000kg)
TBD	To be determined
Tc	Chamber temperature
TCS	Thermal control system
TEI	Trans-Earth injection
TEIS	Trans-Earth injection stage
t.f.	Tank weight factor
THC	Temperature and humidity control
TMI	Trans-Mars injection
TMIS	Trans-Mars injection stage
TPS	Thermal protection system
TT&C	Tracking, telemetry, and control
T/W	Thrust to weight ratio
UN-W/25Re	Uranium nitride - Tungsten/25% Rhenium reactor fuel
VAB	Vehicle Assembly Building
VCS	Vapor coolled shield
Vinf	Velocity at infinity

WBe_2C/B_4C	Tungsten beryllium cabide/Boron cabide composite
WMS	Waste management system
W/O	Without
WP-01	Work package 1 (of SSF)
w/sq cm	Watts per square centimeter (should be Wcm^{-2})
Z	Atomic number
zero g	An unaccelerated frame of reference, free-fall

[order: numbers followed by greek letters]

100K	$\leq$100,000 particles per cubic meter larger than 0.5 micron in diameter
7*n*7	Where n=(0,2-6): Boeing Company jet transport model numbers
°k	Kelvin (K)
+e	Positive charge equal to charge on electron
-e	Charge on electron
ΔV	Change in velocity
s	Standard deviation
μg	Microgravity (also called zero-gravity)

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I. Evolution of the Concept

A. Reference Concept Development

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I. Evolution of the Concept

During the *90 Day Study*, NASA's two Office of Space Flight (Code M) Space Transfer Vehicle (STV) contractors supported development of SEI lunar transportation concepts. This work treated lunar SEI missions (and evolution to the support of later Mars missions) as the far end of a more near-term STV program, most of whose missions were satellite delivery and servicing requirements derived from Civil Needs Data Base (CNDB) projections. STCAEM's contribution to that effort focused mainly on crew system design, since this was recognized as offering potential for commonality with crew cab design for Mars excursion vehicles (MEVs).

Later, STCAEM began to address the complete design of a lunar transportation system. Because of our Mars concept experience, our perspective was particularly sensitive to evolutionary systems; the approach of looking *back* from a Mars mission perspective is thus complementary to that of the parallel NASA studies. Our effort was guided by attention to two broad drivers. First were precisely those technical requirements whose resolution had proved so intractable for earlier concepts:

1) State-of-the-art understanding of constraints imposed by the detailed **geometry of aerobraking** upon Earth return: non-symmetrical relative wind configurations for lifting flight profiles; off-axis placement of composite mass-center (CM); and changing mass-balance conditions due to sequential propellant expulsion.

2) The need to accommodate **"mixed" payloads** in a reasonable lunar exploration program: versatility in the delivery of a wide variety of heavy-cargo payload manifests, rarely if ever mass-split evenly; cargo processing and loading requirements in LEO; cargo exchange between transfer vehicles and excursion vehicles; cargo offloading on the surface of the Moon; cargo placement on manned flights; and shirtsleeve (IVA) exchange of crew between transfer and excursion vehicles.

3) Provision for **transfer of cryogenic propellants**: a typical scenario is supplying LH_2, brought from Earth by a transfer vehicle, to a reusable lander based in low lunar orbit (LLO). Cryogenic propellants are baselined, of course, because of the requirement for high-thrust propulsion for planetary landing and ascent. (The use of nuclear thermal propulsion for lunar transfer is potentially attractive, but still involves cryogenic propellant management.)

4) Potential for **full system reusability**: designs which drop tanks are better for limiting aerobrake size, but have negative cost implications for advanced cryogenic storage technology, and

negative operational implications via the accumulation of empty tanks in cis-lunar space and on the lunar surface. Mission modes which posit multiple annual flights for several decades drive us to consider full reusability.

Second, we recognized that over several decades of lunar operations, many mission modes should be accommodated. What is needed is not so much a single vehicle or pair of vehicles, but rather an evolving lineage of vehicles, fabricated on long-lived production lines, which can be adapted gracefully and economically to handle contemporary requirements. Two observations keyed this investigation:

1) Lunar flight hardware decisions will probably be made before final site selection decisions. This means that the lunar transportation architecture should be careful not to constrain site selection to less than potentially global access. Many possible mission modes must be preserved by the architecture.

2) An early lunar surface operations capability can be obtained by using a tandem-direct flight mode, in which one lunar transfer vehicle (LTV) boosts another, "campsite" LTV, to a fractional-orbit, direct landing on the Moon. The crew would be sent separately on an identical profile, returning directly to Earth's surface in their heat-shielded crew capsule. No LLO operations, no aerobrake, no LTV recovery, and no space station rendezvous upon return would be needed, *nor would a specialized lunar lander (LEV).*

What resulted was a lunar transportation family (LTF) concept, consisting of various "models" of two basic, cryogenic vehicle "chassis": an LTV with 110 t propellant capacity, and an LEV with 25 t capacity. Particular vehicle combinations from this evolutionary family can handle 11 distinct mission modes, to provide versatile, flexible service for decades as mission requirements evolve. For instance, the addition of an aerobrake would permit unmanned recovery of the boost-stage LTV, providing invaluable flight qualification experience for later man-rating. (Such an aerobrake can be essentially the symmetrical central core of the asymmetrical Mars-class aerobrakes discussed later, since the L/D requirement is only about 0.25 for lunar missions; aerobrake technology evolution is then enhanced.) A heavy-cargo lander would be a modest upgrade to the campsite vehicle design. If the scale of the exploration architecture justified the more efficient lunar-orbit-rendezvous (LOR) mode, a dedicated lunar excursion vehicle (LEV) could be introduced. For fully reusable operations, a version of the campsite habitation module would provide crew support during the extended-wait required by return orbit phasing. If electric propulsion Mars missions were operated efficiently through a lunar libration-point, LTVs could support these as well as lunar operations. And LTVs could supply the final capture propellant to an NTR vehicle returning from Mars. Lunar transportation operations can be upgraded to the use of lunar-derived oxygen (LLOX) with this family of vehicles also. All combinations of crew and cargo manifests identified so far for lunar support, and all lunar-related SEI missions identified so far, can be accommodated by the LTF.

LTV/LEV Concept Constraints

Past industry-wide concept development efforts for lunar transportation systems have identified several complicating factors, listed and elaborated here. Successfully accommodating any or all of them severely constrains the configuration options available. The STCAEM Study has adopted all five as design requirements. As they get incorporated into defined vehicle concepts, they will be recorded as design-verified performance requirements.

LTV / LEV Concept Constraints

STCAEM concept development directly addresses 5 challenges

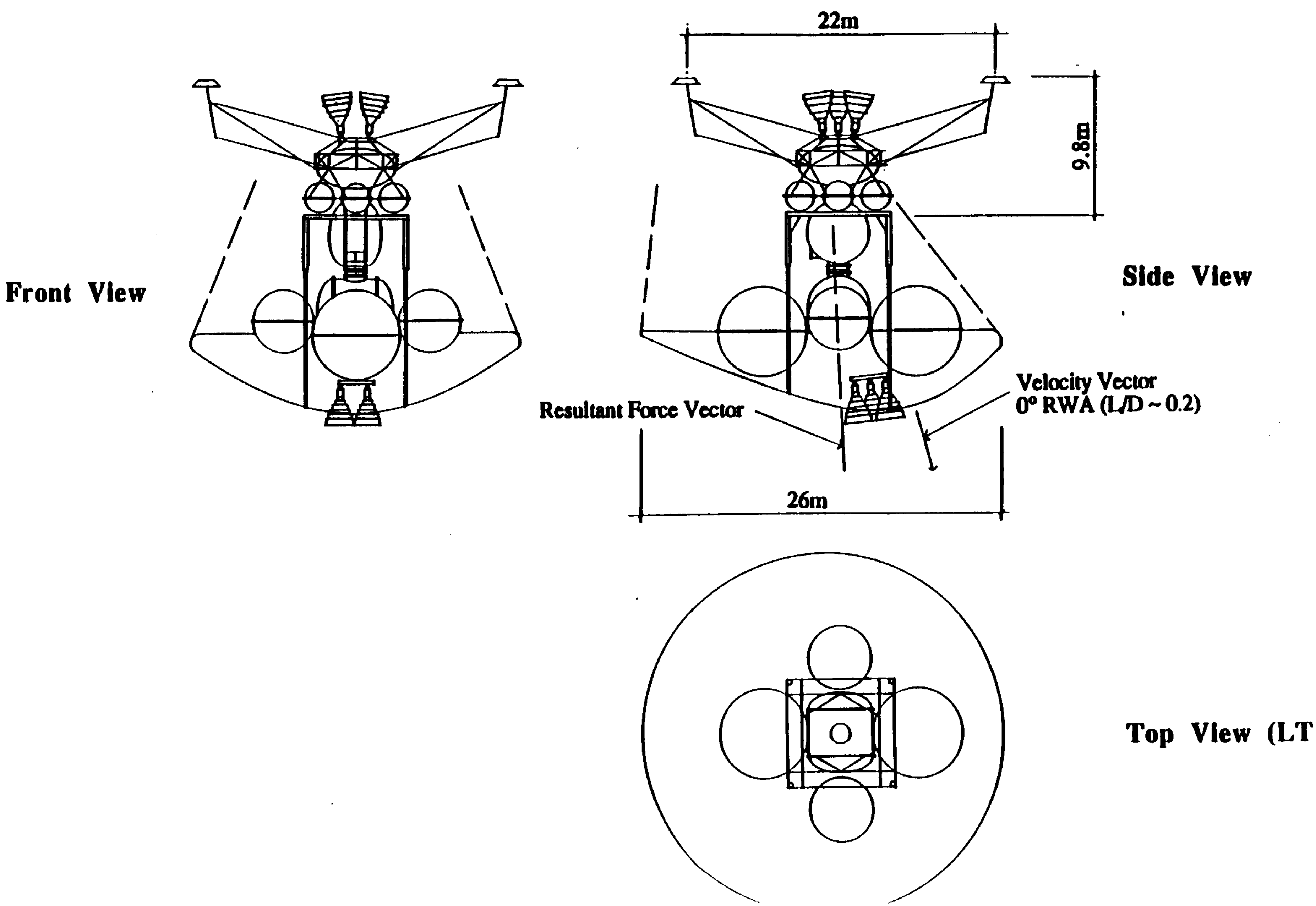
- **The geometry of aerobraking** (non-symmetrical relative wind configuration, proper vehicle CM placement, changing mass-balance conditions)
- **Accommodation of mixed payloads** (versatile cargo manifest delivery, transfer between vehicle stages, and processing at SSF)
- **Cryogenic propellant transfer in LLO** (Options: slow-spin pumped; closed thermodynamic with vented chilldown; open-vent quiescent fill)
- **Fully re-usable design** (no drop-tanks; potentially >5 flights per vehicle)
- **Growth capability** (ganged LTVs for evolutionary architectures, higher energy or alternative missions)

LTV/LEV Configuration

This and the next chart sketch configuration concepts for the cis-lunar system which uses the common parameters just developed. Sized for the Mars case, the aerobrake is somewhat larger than strictly required by the lunar mission, despite retaining all propellant tanks throughout the mission profile. Furthermore, it has the higher-L/D shape required for Mars, albeit flown here in a lower L/D attitude. The LTV engines, while oriented to accommodate the vehicle's changing mass center, are positioned according to Mars landing requirements. Direct transfer of crew from LTV to LEV is accomplished in the same configuration as propellant transfer (whether pumped with rotational settling or transferred using a μg technique).

A single, unconstrained payload pallet is transferred at this time also. The pallet can be integrated at SSF separately from the LTV processing, then mounted for TLI, transferred to the LEV, and unloaded on the surface by a straddling payload transporter. The LEV's height is reduced as much as possible, given the constraint of engine-out on ascent, to maximize unloading efficiency. The landing gear would permit settling the LEV lower to the ground after touchdown to facilitate unloading as well, and are configured in plan to accommodate a triangular straddler. The center section of the pallet is removable, and passes over the LEV-mounted crew module for cargo transfer during crew missions (heavy, bulky, singular payloads like habitat modules or process reactors cannot be accommodated on crew flights due to mass capacity considerations anyway), allowing manifesting of resupply cargo. For unmanned cargo flights, the full payload pallet would be used. The pallet retracts close to the LTV tanks for the aeromaneuver upon return to Earth.

LTV/LEV Configuration



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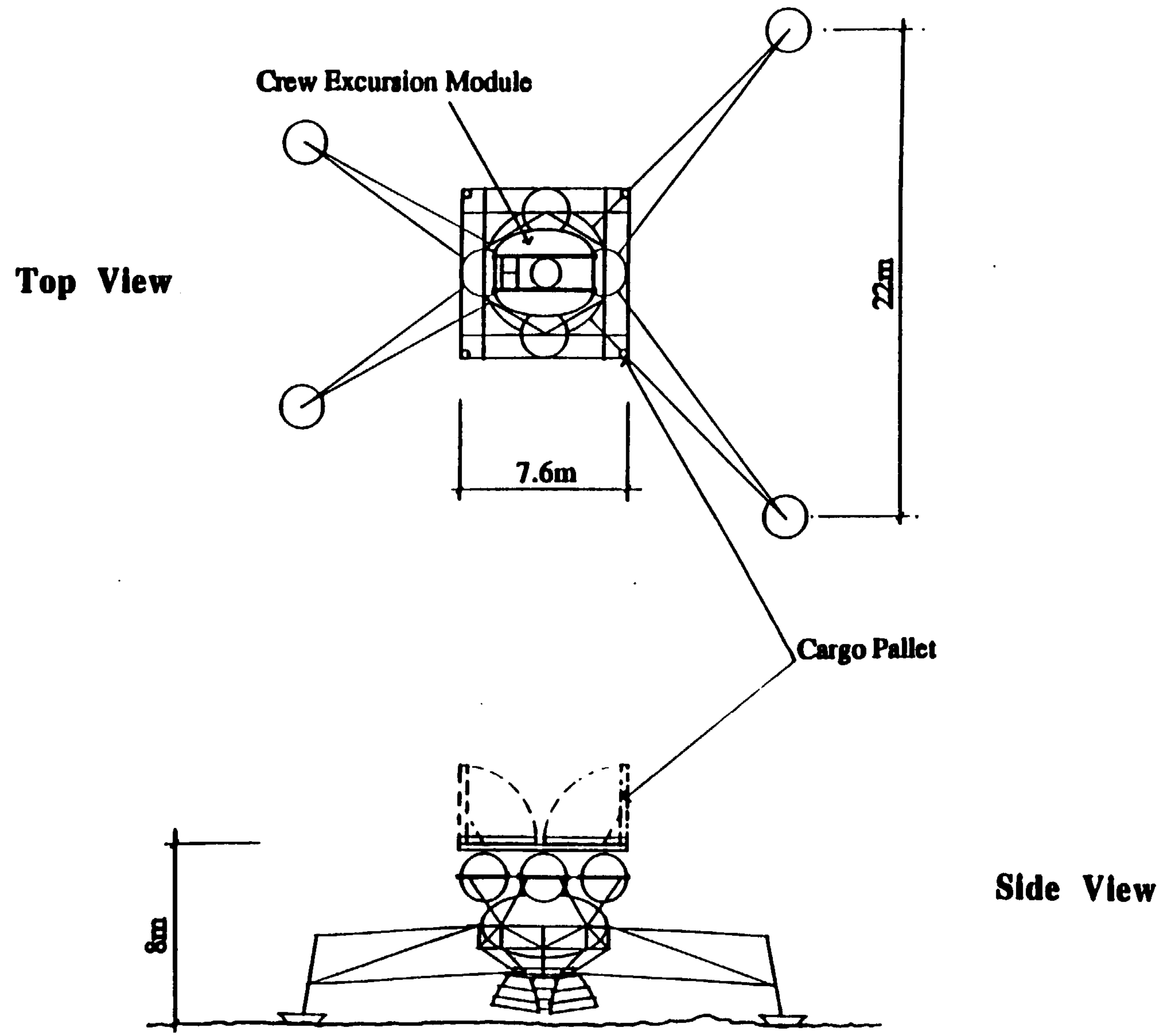
LEV Surface Configuration



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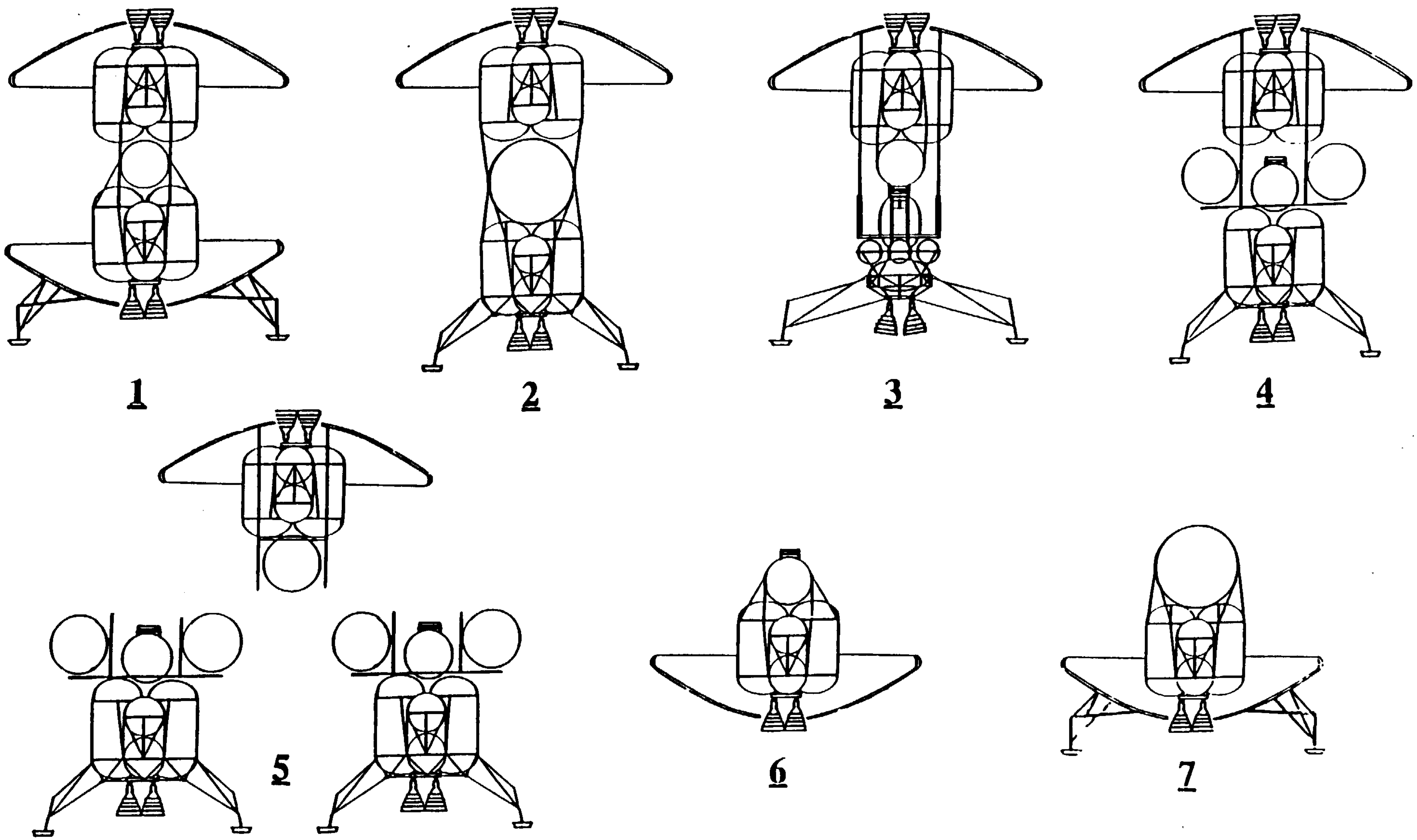
Lunar Vehicle Configurations

Seven lunar vehicle configurations are shown below, to correspond to the mission modes depicted on the previous pages. The vehicles shown are based on a "kit of parts" that include the 26 m aerobrake, a four crew cab, six crew transit hab, 110 t propellant and engine combination, 25 t propellant and engine combination, and a standard cargo pallet.

Lunar Vehicle Configurations



1 2 3 4 5 6 7

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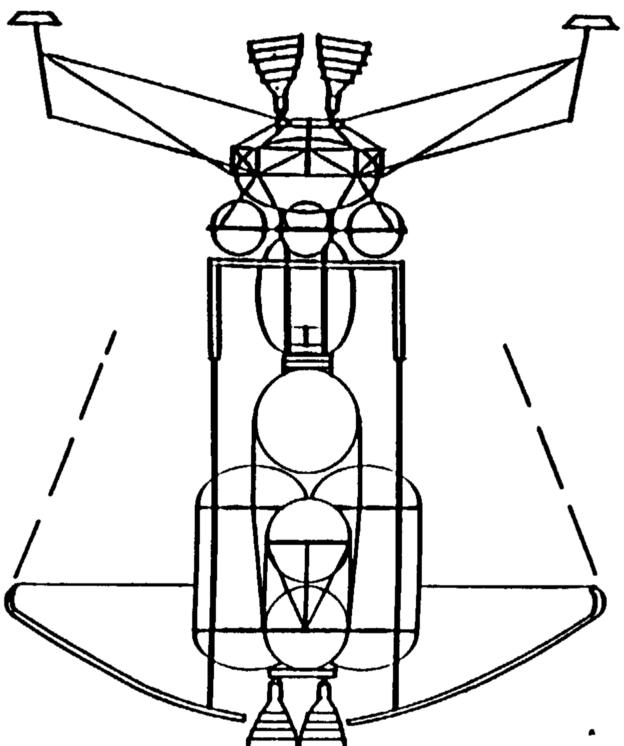
LEV / LTV Configurations

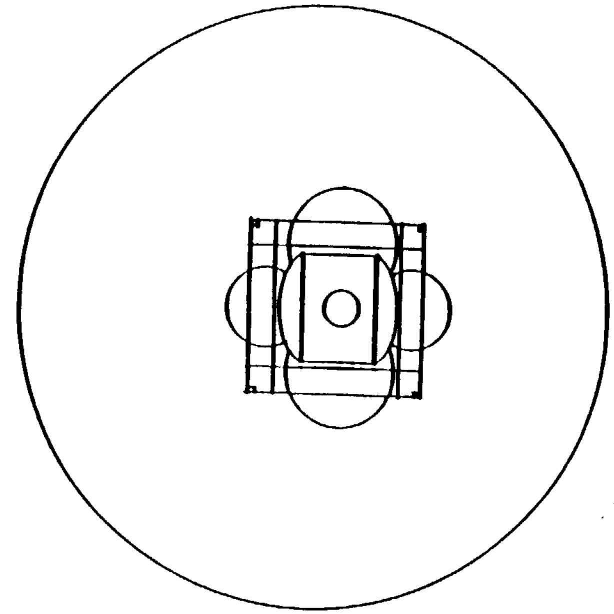
Shown below are configurations for the LEV and LTV, based on a 110 t propellant load and a 25 t propellant load. Both configurations are capable of being launched intact within a 10 m. payload shroud. The LTV is shown with the 26 m aerobrake, configured for an L/D of 0.5, and a relative wind angle of 0 deg.

LEV / LTV Configuration

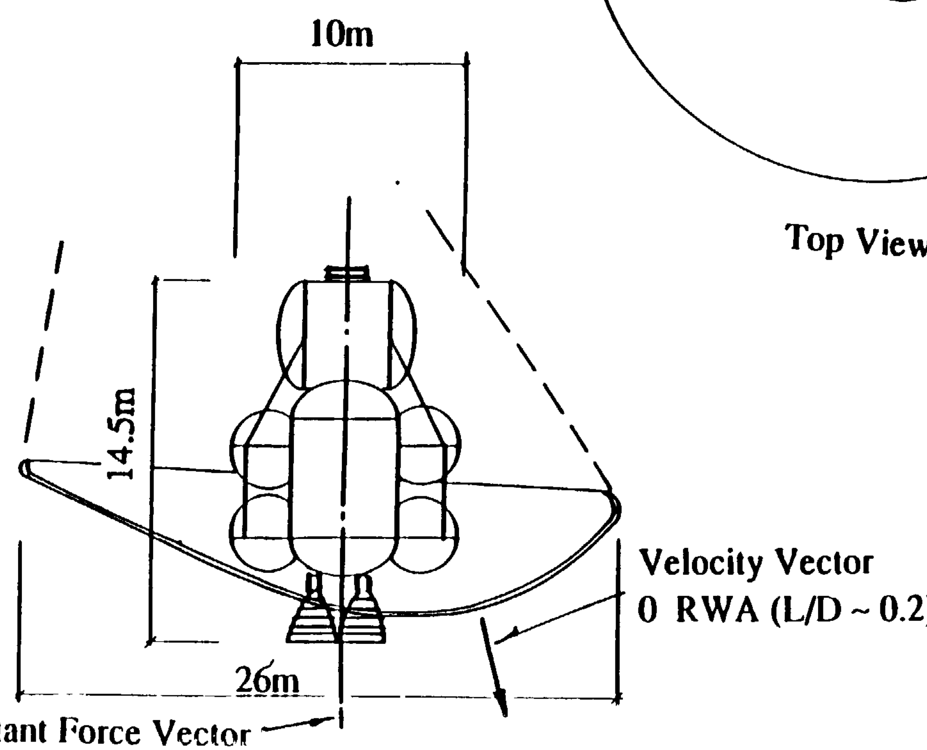


Front View

- 110 t propellant in LTV
- 25 t propellant in LEV
- Packages in 10 m shroud
- Uses mini-MEV aerobrake



Top View



Side View

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Lunar Transportation Family (LTF)

Threefold Design Strategy

Tough Drivers: - Current understanding of aerobraking geometry

- Mixed payload manifests, exchange operations, offloading (cargo & crew)
- Cryogen transfer
- Re-usability (retaining tanks also)
- Subsystem/component commonality for later Mars vehicles

Multiple Mission Modes

- Transportation infrastructure decisions will precede site selection, and should not constrain it
- Global lunar access must be preserved

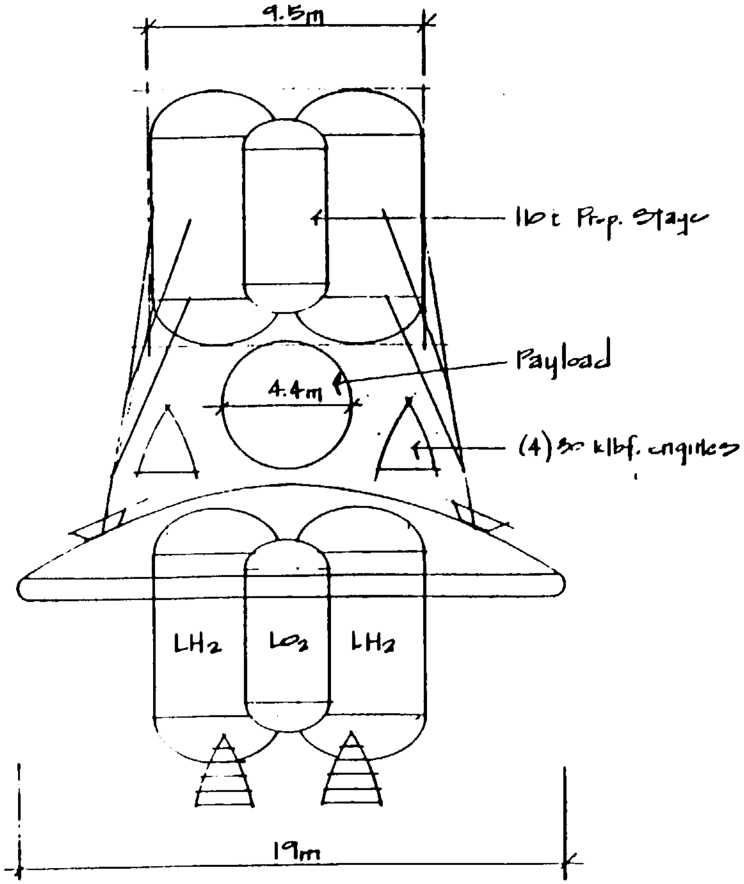
Early Capability: "Campsite" operations (expandable mode, without aerobrake or LEV)

Lunar Family Configuration 1, 2 & 3

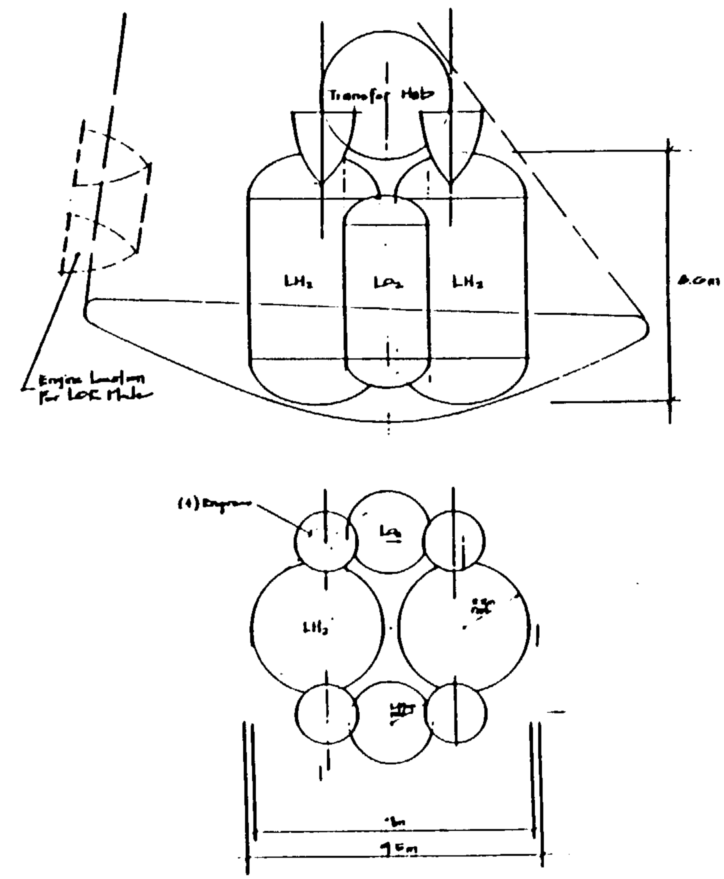
The following 3 charts show preliminary sketches of the latest phase of what is now being called the Lunar Transportation Family. The concept is to develop a few standard components that can be pieced together to satisfy the requirements of various mission modes. These sketches show a few of the missions that can be accomplished with these components.

Lunar Family Configuration (1)

Tandem Direct
Campsite, Crew, or Light Cargo

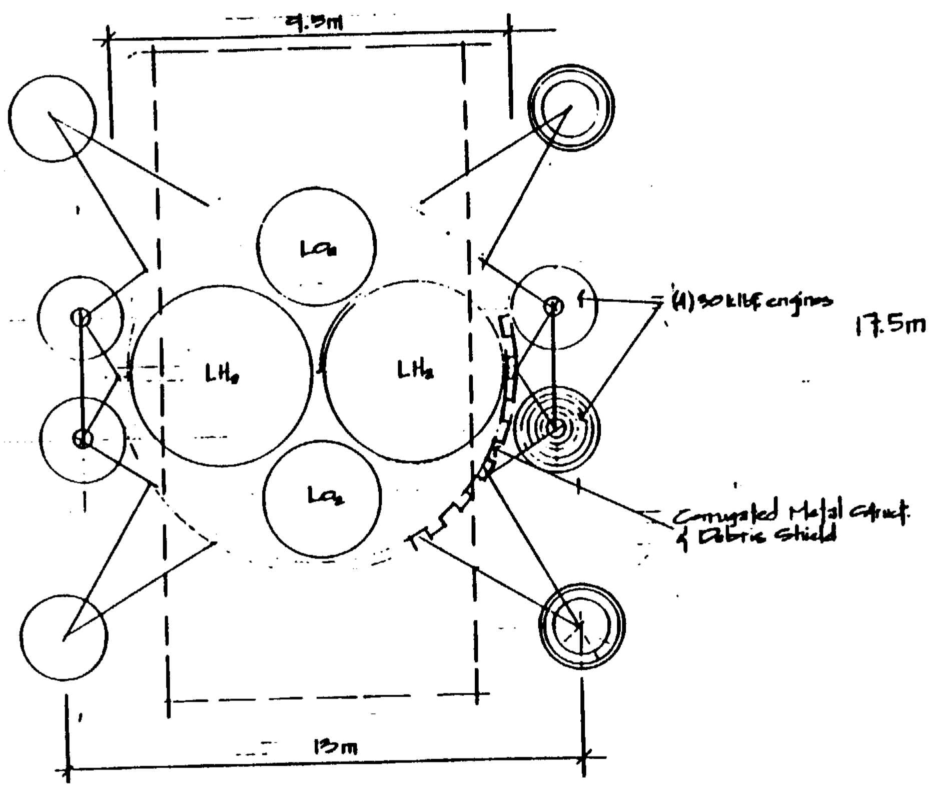


Transfer Vehicle
Tandem or LOR

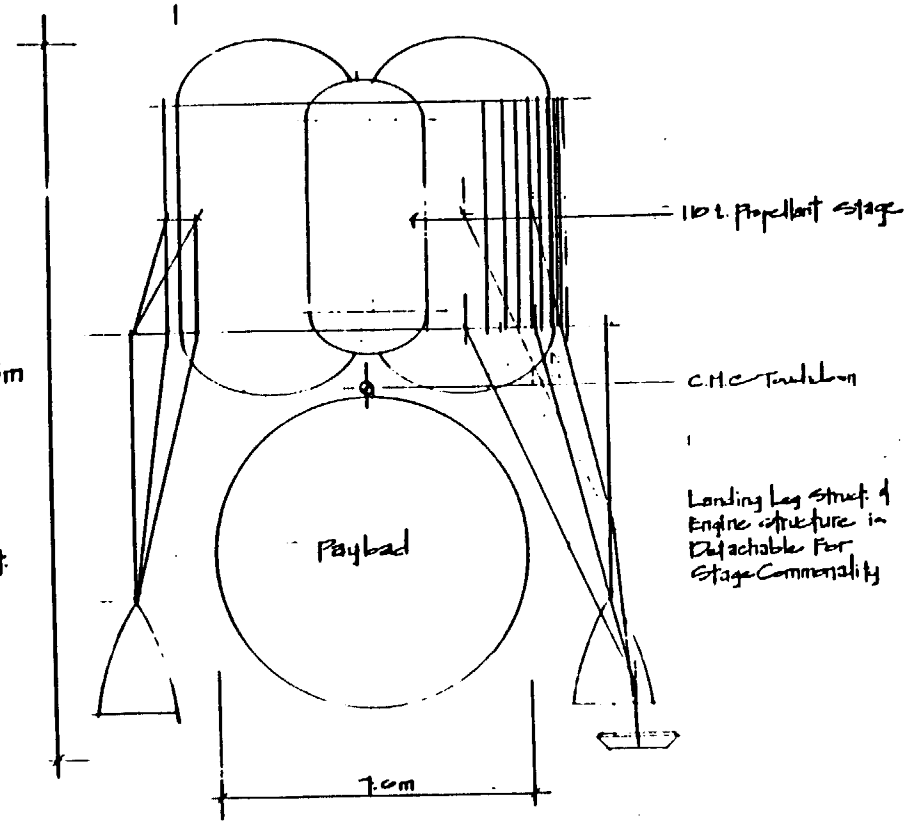


Lunar Family Configuration (2)

Tandem Direct Heavy Cargo Lander



Top View



Front View

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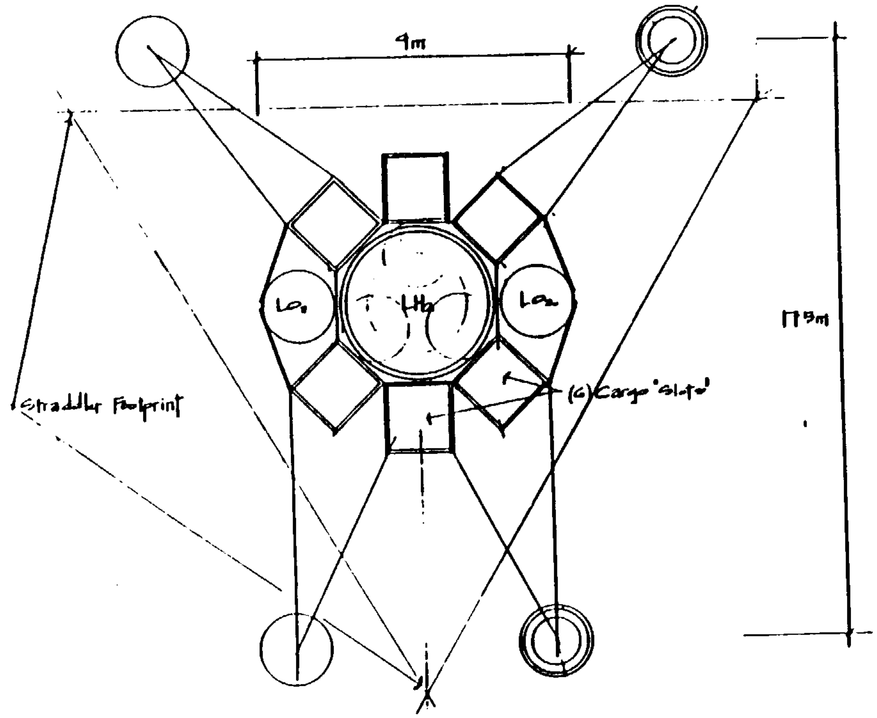
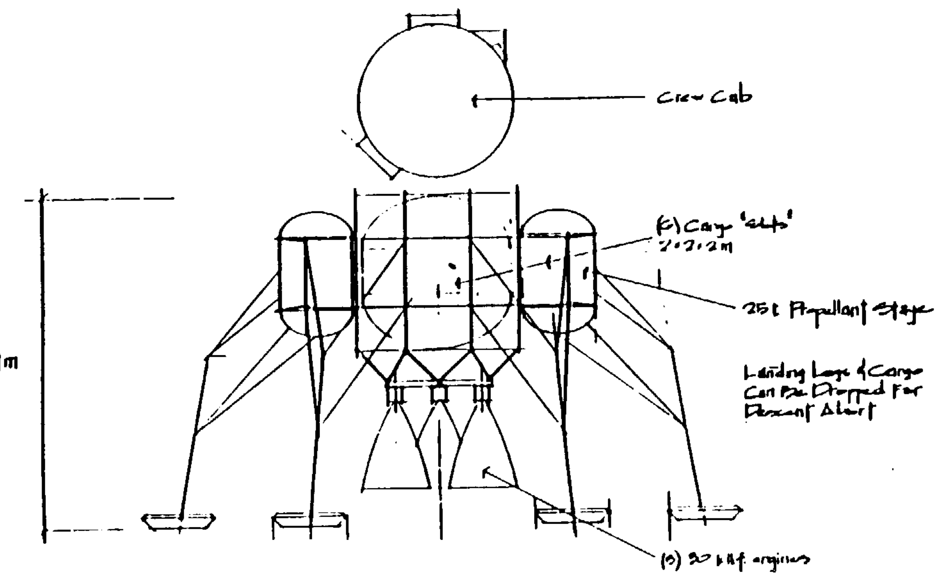
Lunar Family Configuration (3)

BOEING

Lunar Excursion Vehicle



Top View



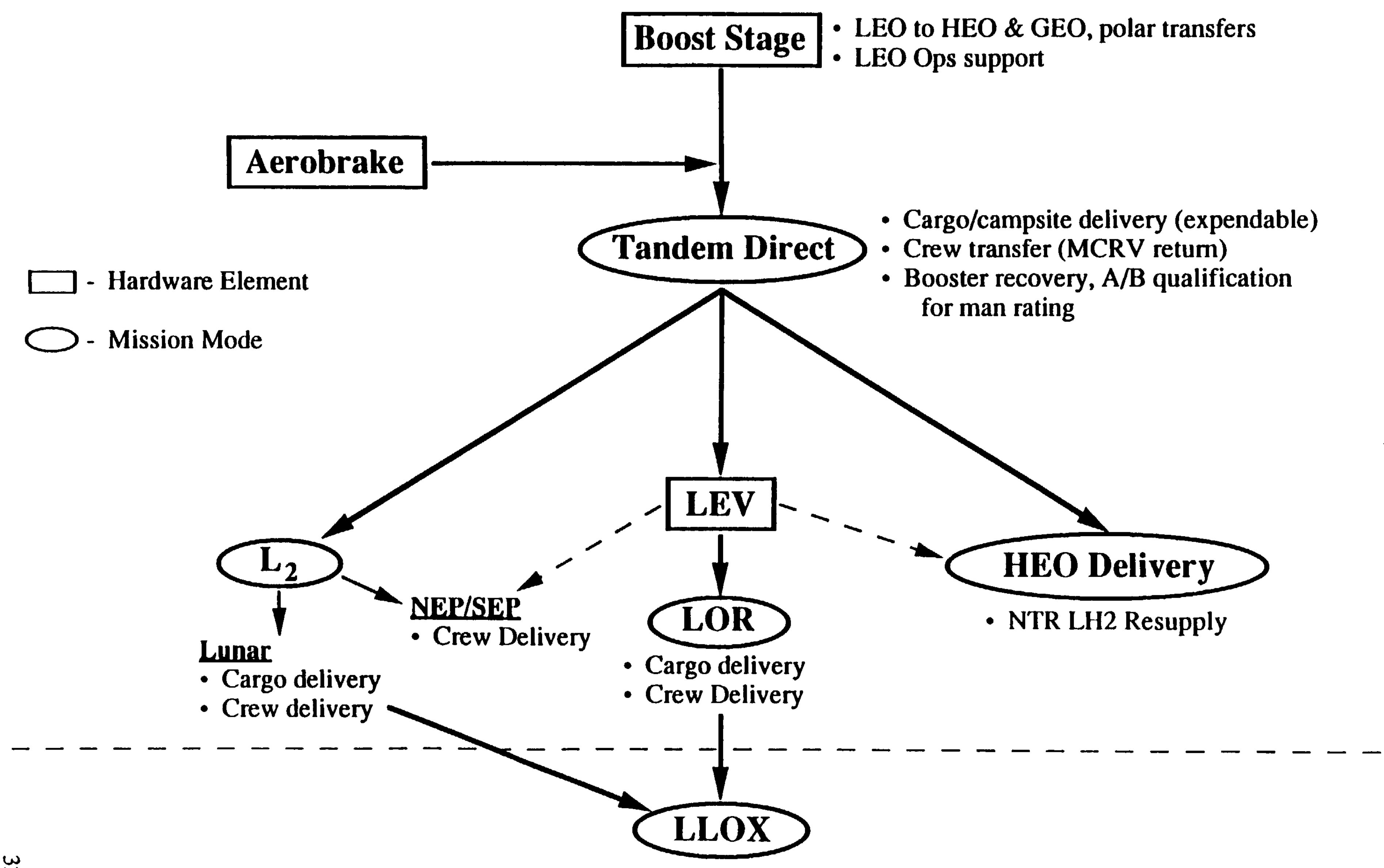
Front View

Lunar Transportation Family (LTF)
Preferred Evolution

The LTF concept evolves by beginning with minimal hardware development to phase the costs over the duration of the program. **First**, a boost stage is developed for Earth orbit-to-orbit transfers and operations support in LEO. Completely expendable tandem-direct missions are flown to the lunar surface at this point using an MCRV for crew return. **Secondly**, when the aerobrake comes on-line, tandem-direct missions which recover the boost stage are flown for aerobrake flight qualification for man-rating. Using the same hardware, L_2 missions are flown for crew delivery to NEP/SEP vehicles as well as LOR lunar crew and cargo delivery. NTR LH_2 resupply missions can also be flown at this point to provide propellant to get the NTR to SSF orbit for refurbishment. **Third**, an LEV is brought on line so that classical LOR missions are possible to deliver both crew and cargo to the surface. **Fourth**, LLOX usage becomes available and the missions to the lunar surface begin to take advantage of surface refueling to reduce Earth-to-orbit transfers of propellant.

Lunar Transportation Family

Preferred Evolution



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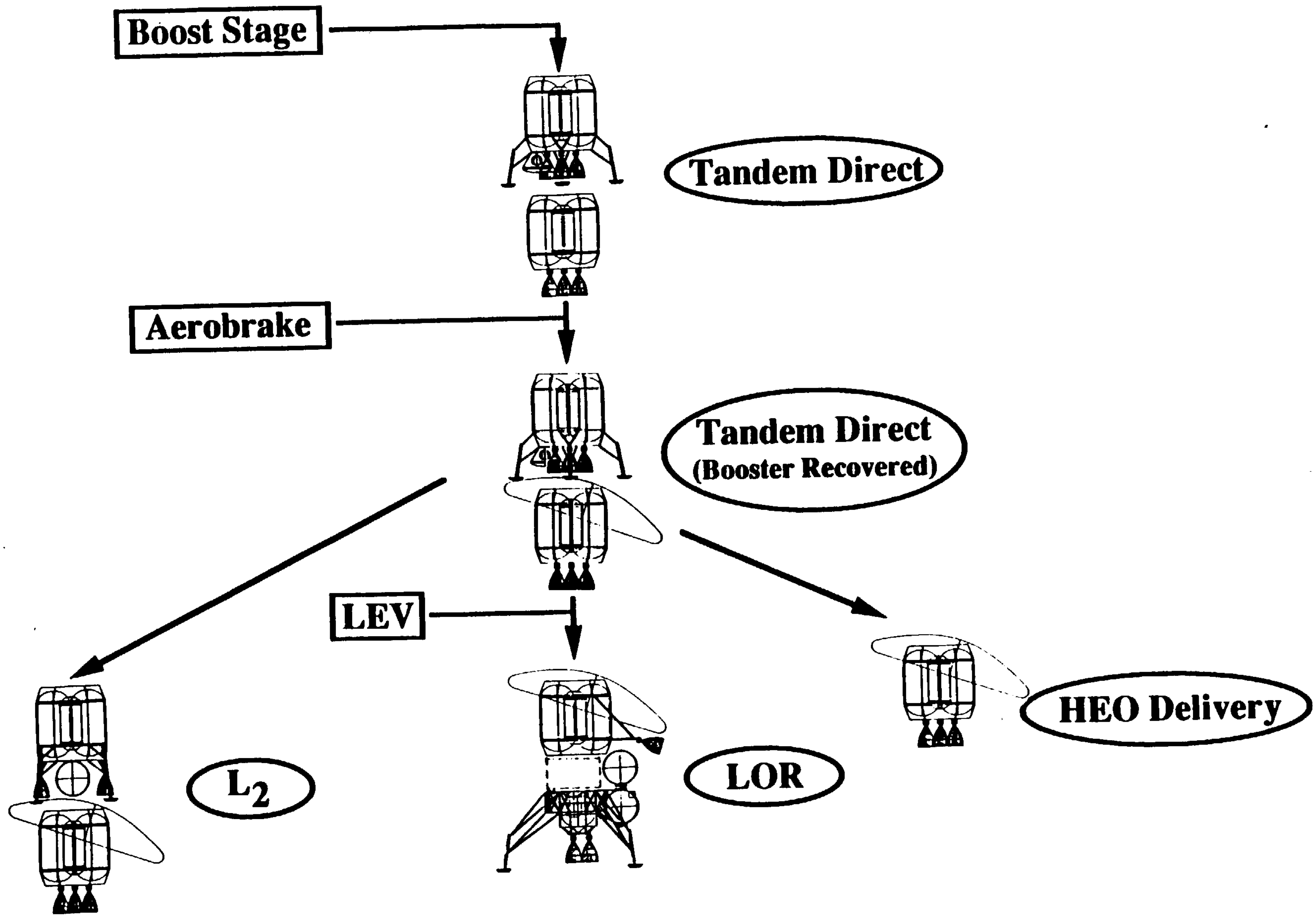
Lunar Transportation Family
Evolution



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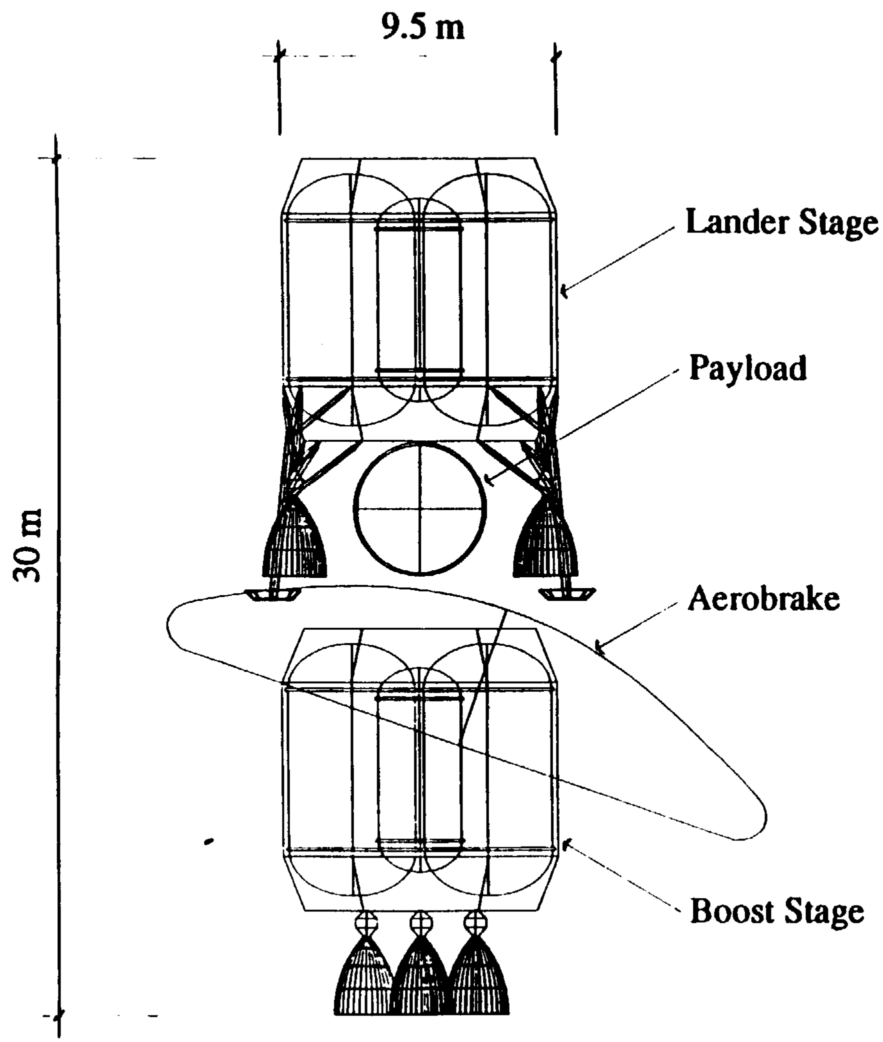
Lunar Transportation Family Configurations

The next 11 charts show configurations for the evolution of the lunar transportation family with accompanying mass statements. 5 major hardware elements were developed and assembled in different configurations to support 11 different mission modes. This kind of system approach minimizes the major elements required, phases in more complex hardware elements, and provides for commonality throughout the system. Although 11 mission modes are identified and represented, their are other possible missions that can be flown using this hardware, such as the cargo and crew delivery missions through L_2 can be flown with LLOX.

Lunar Transportation Family
Tandem-Direct Cargo

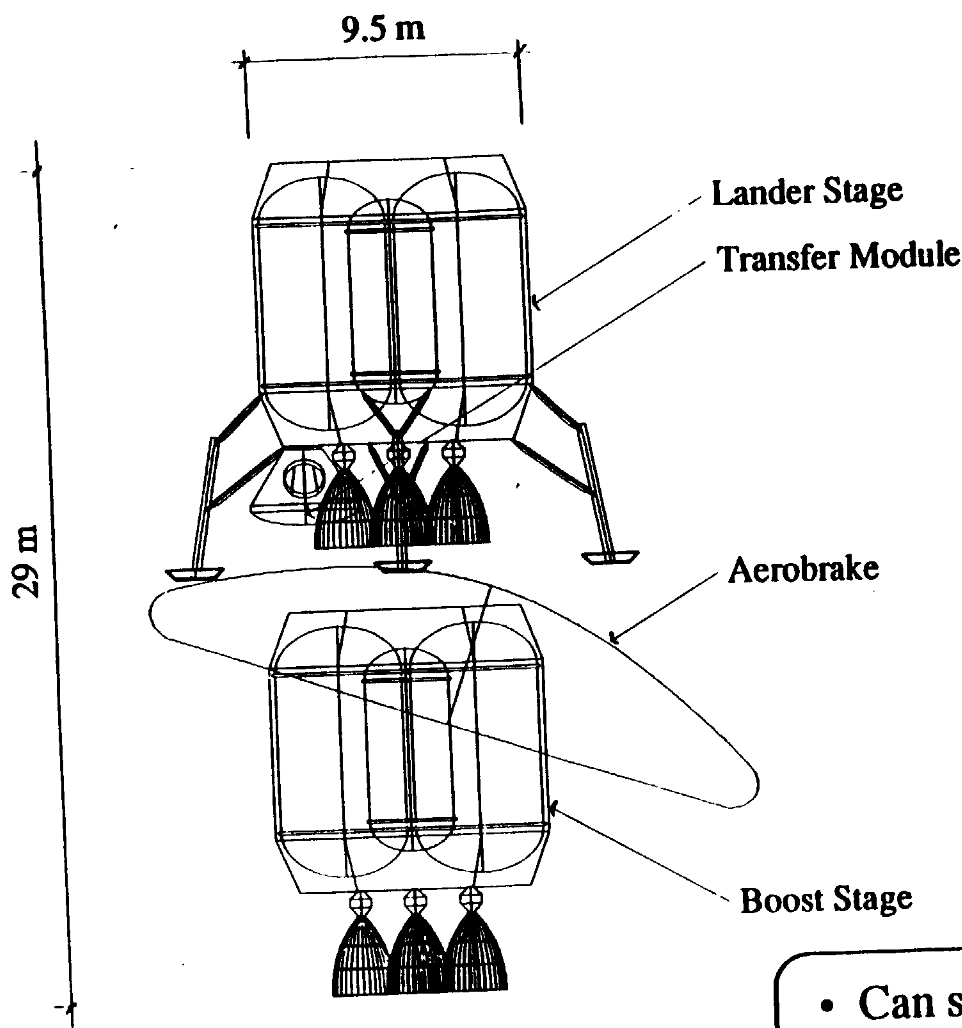


Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	108,950
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	92,603
Payload	60,000
IMLEO (in kg)	**286,304**

- Boost stage flies without an aerobrake initially in an all expendable mode

Lunar Transportation Family
Tandem-Direct Crew

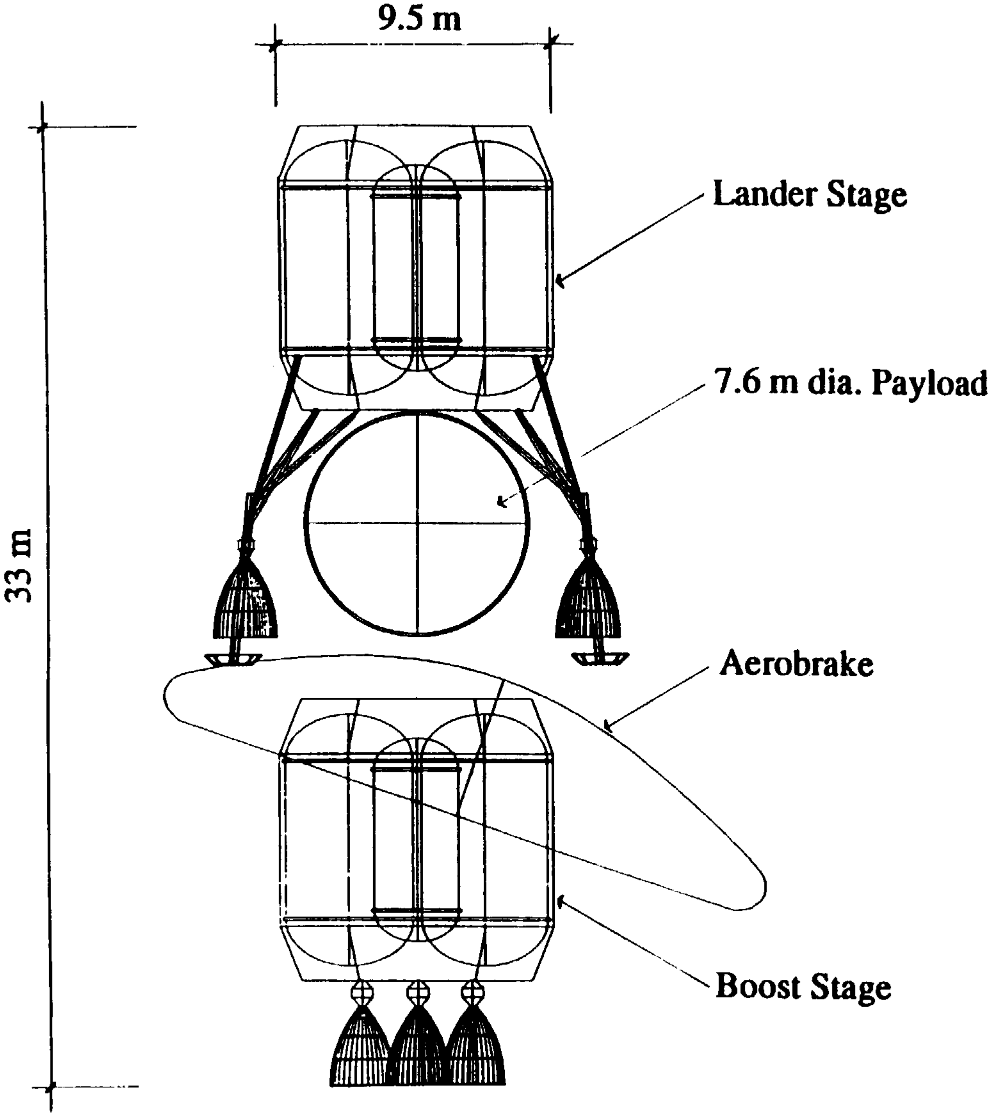


Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	78,647
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	84,436
Transfer Module	13,633
Payload	5,000
IMLEO (in kg)	**206,467**

- Can send up to 8 crew in this mode for crew changeout in early base phases

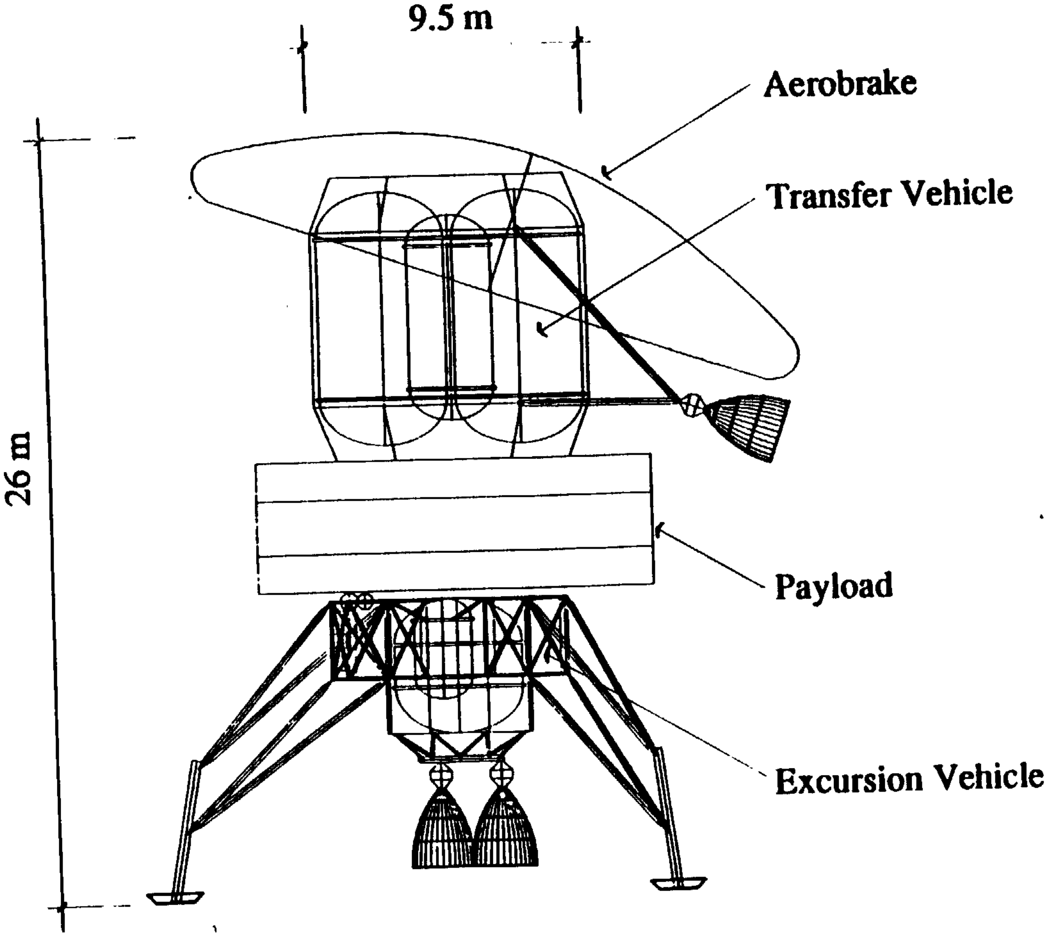
Lunar Transportation Family
Tandem-Direct Large Cargo



Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	108,950
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	92,603
Payload	60,000
IMLEO (in kg)	**286,304**

Lunar Transportation Family
LOR Cargo



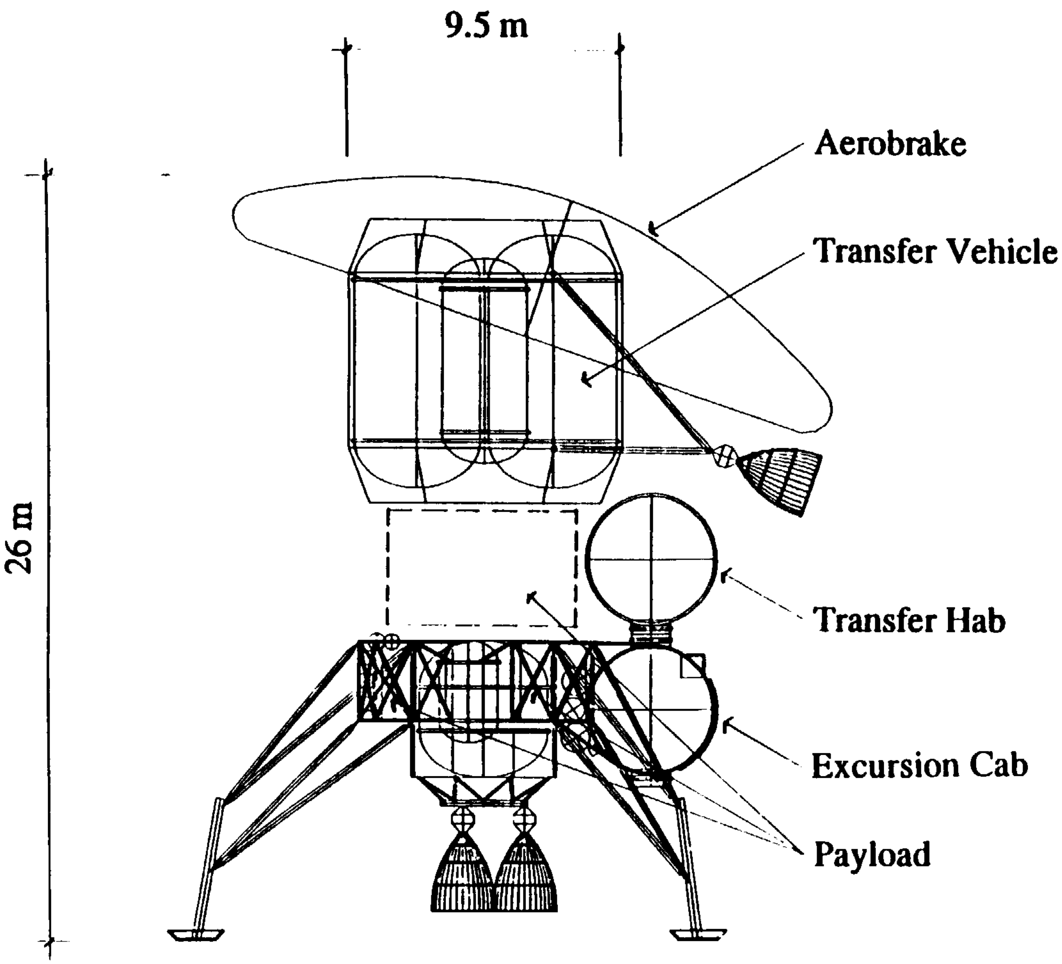
Mass Statement (LEV Delivered)

Transfer Vehicle Inerts	10,638
Transfer Propellant	107,719
Aerobrake	3,000
Excursion Vehicle Inerts	5,592
EV Propellant	23,151
Payload	24,000
IMLEO (in kg)	**174,100**

Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	107,100
Aerobrake	3,000
Resupply Tanks	2,377
Resupply Propellant	23,765
Payload	25,000
IMLEO (in kg)	**171,880**

Lunar Transportation Family
LOR Crew and Cargo



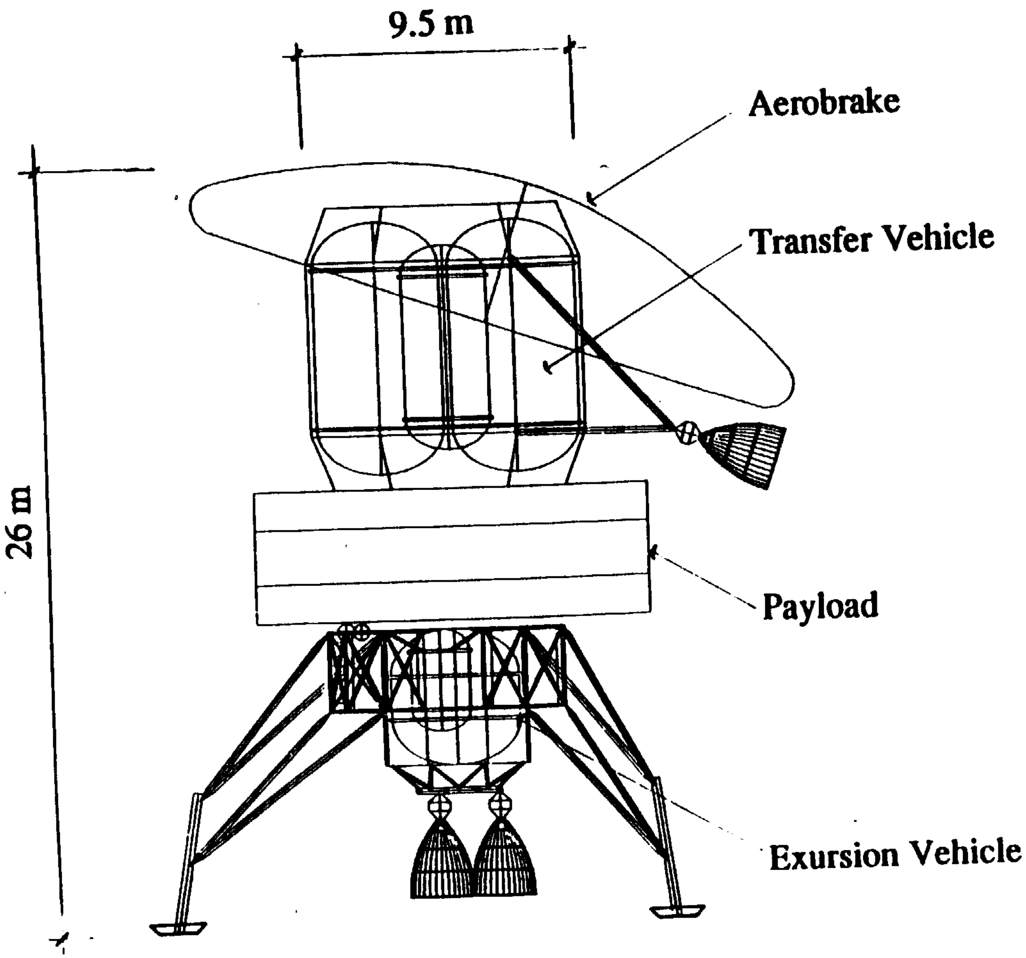
Mass Statement (LEV Delivered)

Transfer Vehicle Inerts	10,638
Transfer Propellant	106,602
Aerobrake	5,000
Transfer Hab	7,500
Excursion Vehicle Inerts	5,592
EV Propellant	19,768
Excursion Cab	4,000
Payload	8,000
IMLEO (in kg)	**167,100**

Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	102,184
Aerobrake	5,000
Transfer Hab	7,500
Resupply Tanks	2,100
Resupply Propellant	20,997
Payload	10,000
IMLEO (in kg)	**158,419**

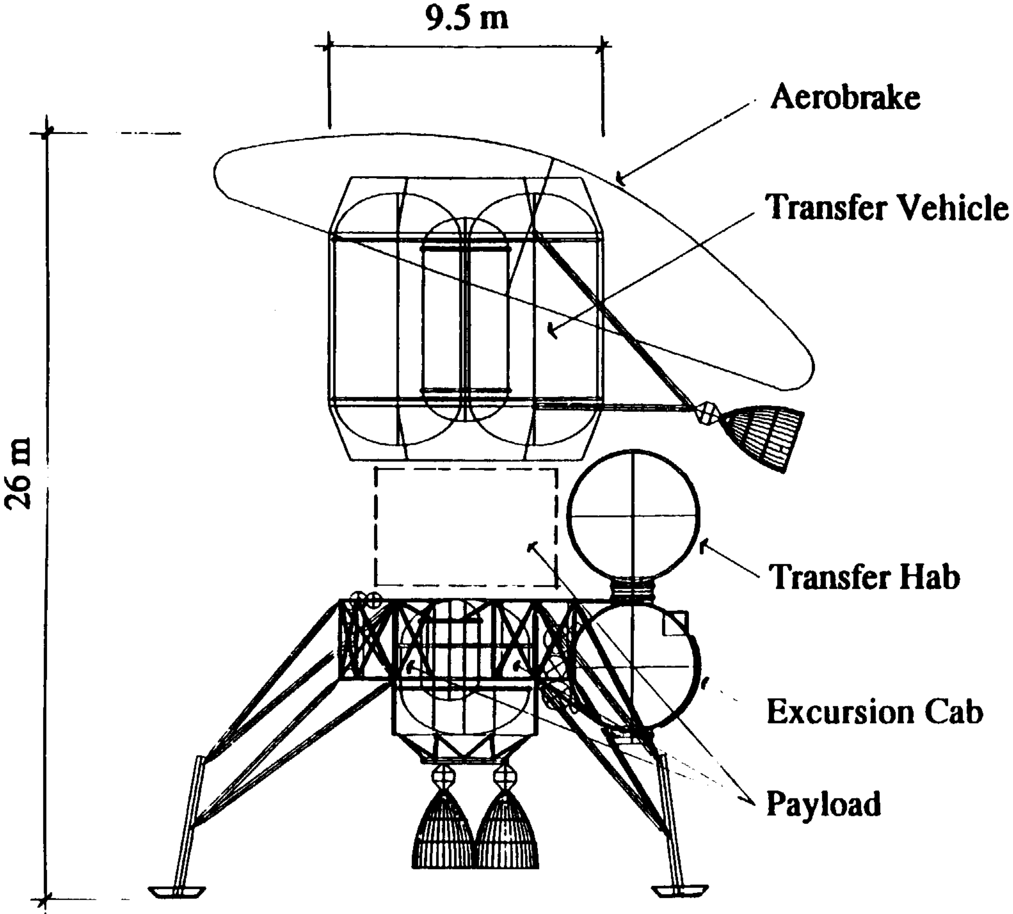
Lunar Transportation Family
LOR Cargo using LLOX



Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	61,645
Aerobrake	3,000
Resupply Tanks	624
Resupply Propellant (LH2)	3,464
Payload	17,000
IMLEO (in kg)	**96,371**

Lunar Transportation Family
LOR Crew using LLOX

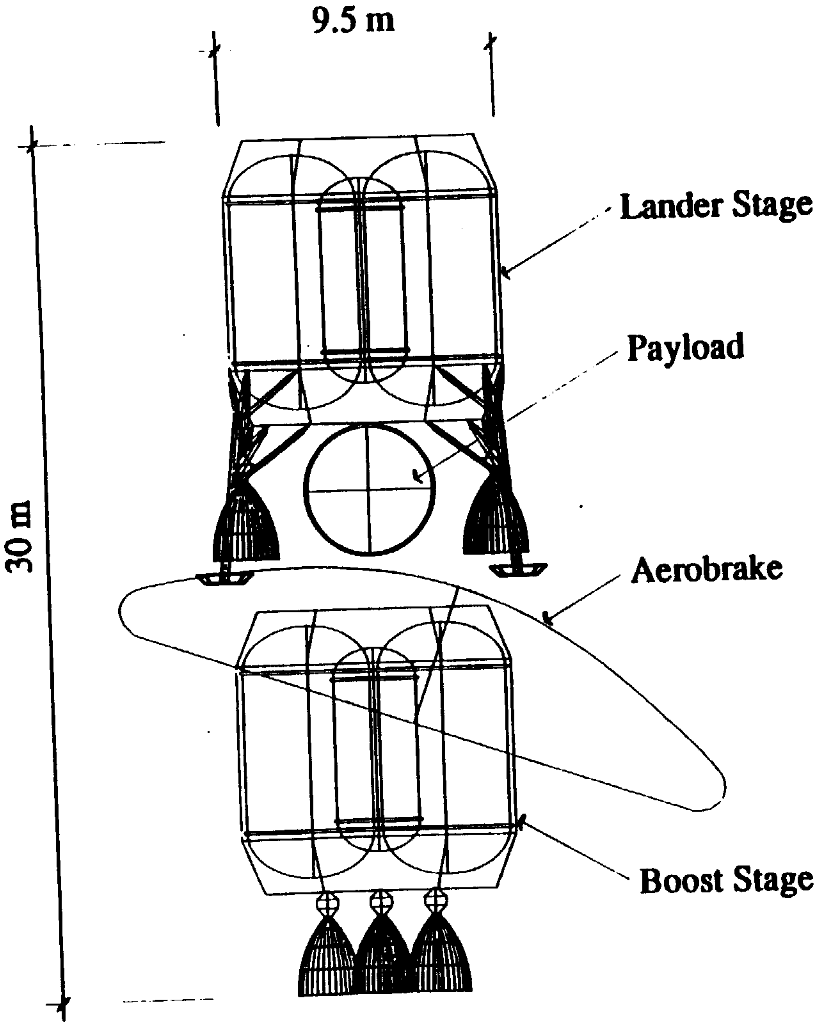


Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	70,067
Aerobrake	5,000
Transfer Hab	7,500
Resupply Tanks	577
Resupply Propellant (LH2)	3,207
Payload	8,000
IMLEO (in kg)	**104,989**

Lunar Transportation Family

Cargo Delivery Through L_2



Mass Statement (Lander Delivered)

Boost Stage Inerts	10,638
Boost Propellant	107,803
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	45,637
Payload	18,000
IMLEO (in kg)	**196,191**

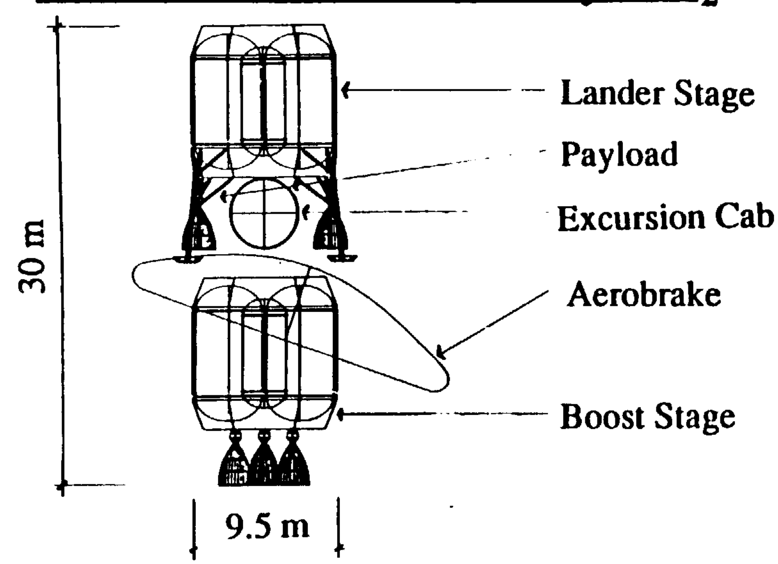
Mass Statement (Lander Refueled)

Boost Stage Inerts	10,638
Boost Propellant	107,734
Aerobrake	3,000
Resupply Propellant	51,419
Payload	24,000
IMLEO (in kg)	**196,791**

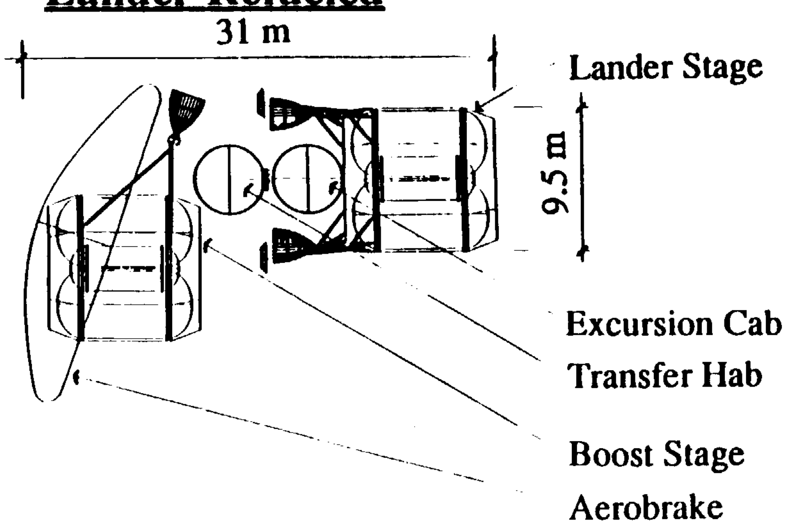
Lunar Transportation Family

Crew Delivery Through L_2

Unmanned Lander Delivery to L_2



Lander Refueled



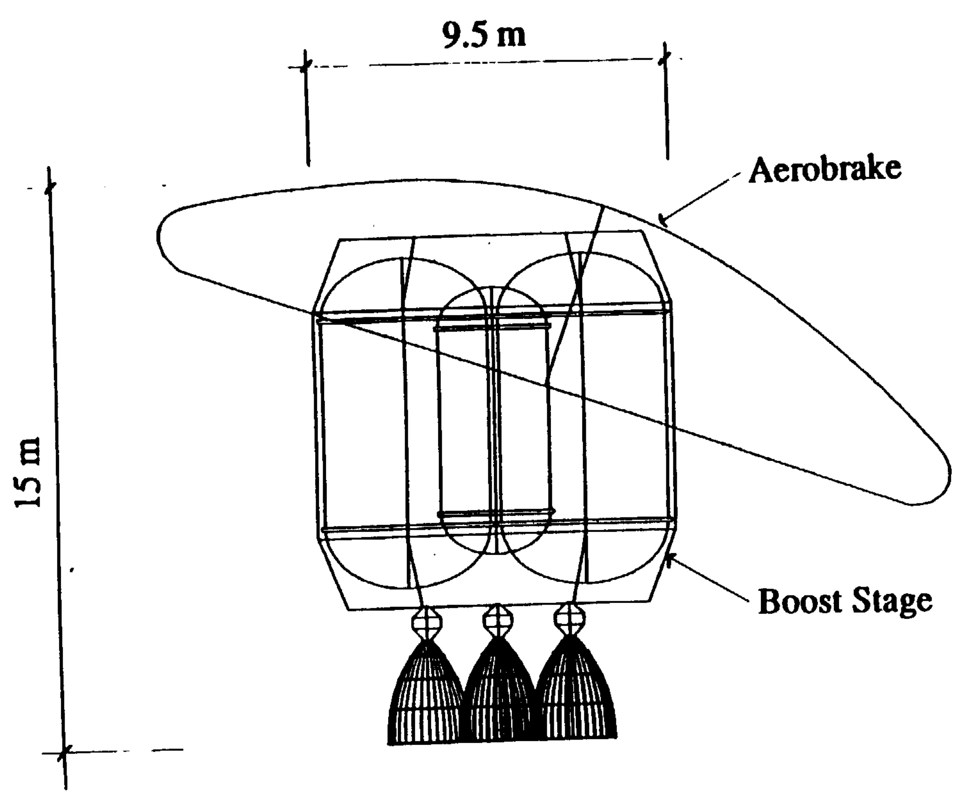
Mass Statement (Lander Delivered)

Boost Stage Inerts	10,638
Boost Propellant	107,877
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	49,415
Excursion Hab	4,000
Payload	10,000
IMLEO (in kg)	**196,043**

Mass Statement (Lander Refueled)

Boost Stage Inerts	10,638
Boost Propellant	107,803
Aerobrake	5,000
Transfer Hab	7,500
Resupply Propellant	51,342
Payload	12,000
IMLEO (in kg)	**194,283**

Lunar Transportation Family
NTR Resupply

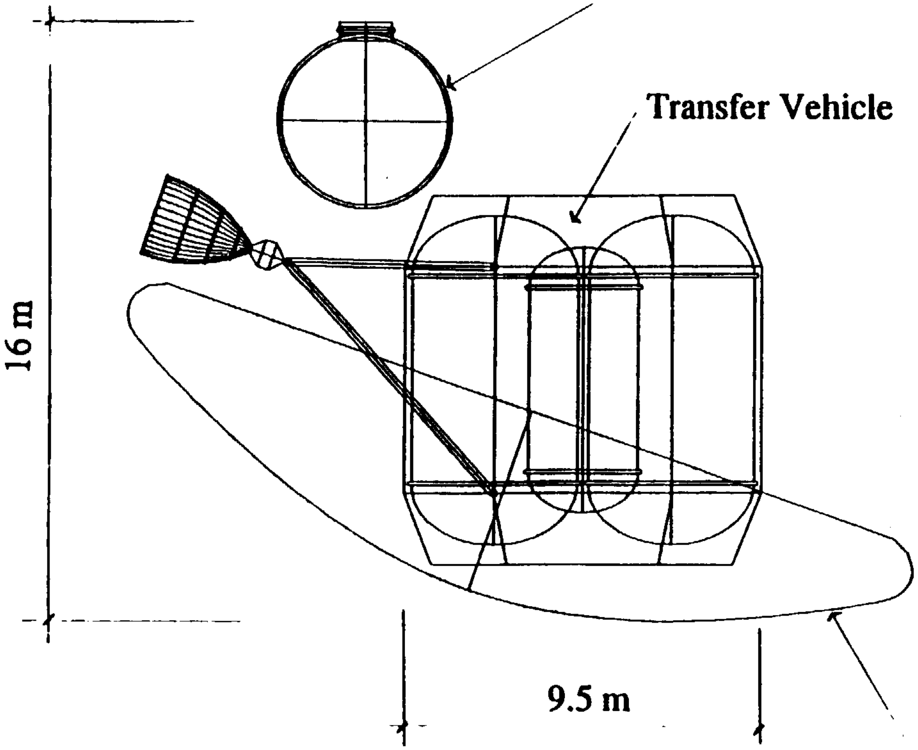


Mass Statement

Boost Stage Inerts	10,638
Outbound Propellant	41,334
Inbound Propellant	687
Aerobrake	5,000
Resupply LH2	30,000
Resupply Tanks	6,600
IMLEO (in kg)	**94,259**

- Delivered resupply propellant is used to transfer NTR to SSF orbit

Lunar Transportation Family
NEP/SEP Crew Delivery



Mass Statement

Boost Stage Inerts	10,638
Outbound Propellant	30,624
Inbound Propellant	2,932
Aerobrake	5,000
Transfer Hab	7,500
IMLEO (in kg)	**56,694**

- Assumes LEO to L_2 transfer & aerobrake in LEO return

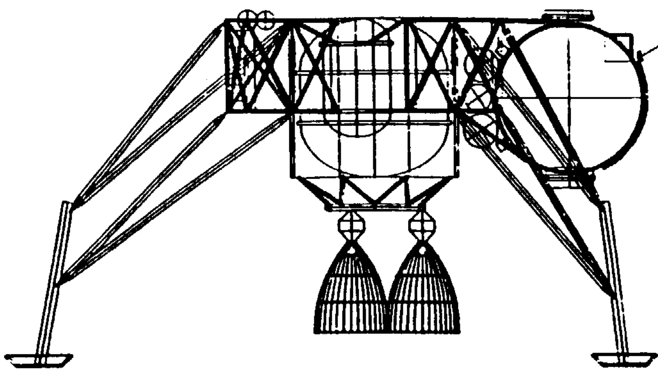
LOR Excursion Vehicle

The LOR Excursion Vehicle shown is a flexible design that can accommodate varying payload sizes, crew + cargo, or cargo only. With the payload imbedded in the lander structure (in standard cargo containers) in the crew mode, this configuration offers the ability to descent abort to LLO by dropping the payload and the landing legs (to have enough propellant to make orbit) and thrusting back to LLO. The placement of the crew cab on the side allows for easy surface access by the crew as well as more direct crew visibility upon landing over top mounted crew cabs. The C.M. shift by placing the crew cab on the side is offset by moving the LO2 off-center and putting the RCS propellant and the avionics on the opposite side of the crew cab. In the cargo-only mode, the crew cab is absent and large cargos can be attached to the top of the lander structure to be unloaded by a straddler. The "triangular" nature of the landing legs is caused by the configuration and size of the straddler, in order for the straddler to easily maneuver over the lander to unload payload.

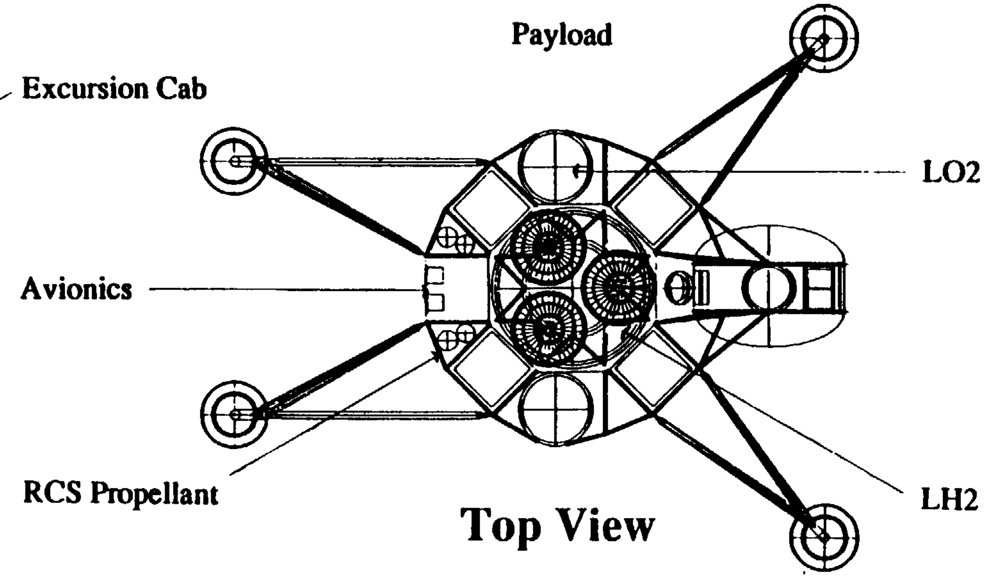
LOR Excursion Vehicle

ADVANCED CIVIL SPACE SYSTEMS

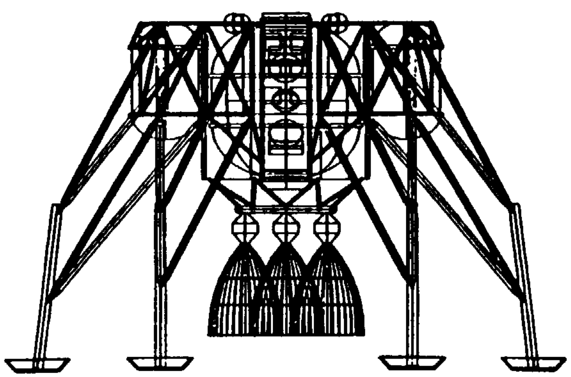
BOEING



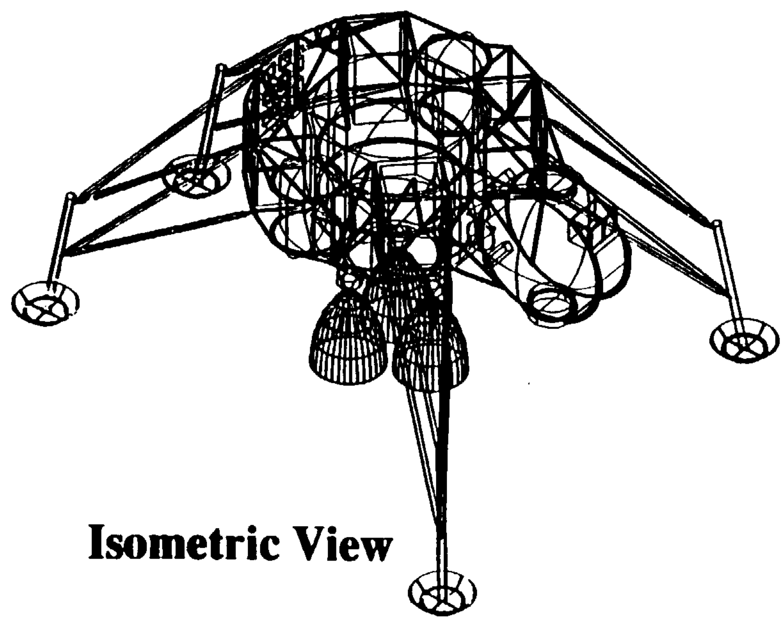
Side View



Top View



Front View



Isometric View

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B. Alternate Concepts

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Preliminary Lunar NTR Vehicles

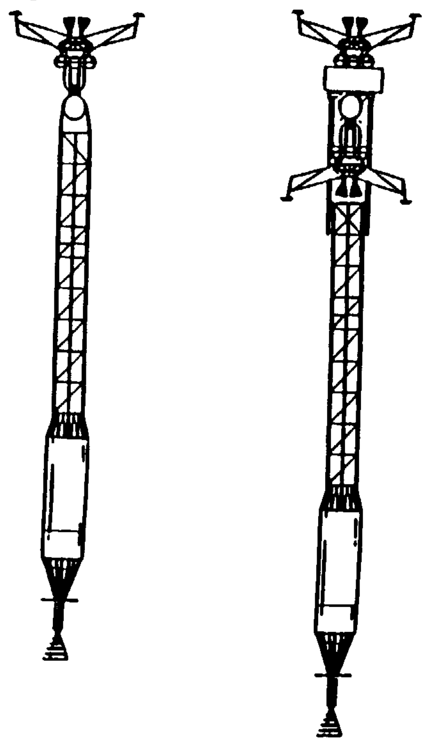
A Lunar NTR vehicle was developed in order to evaluate some of the benefits of NTR propulsion as applied to the Lunar missions. These benefits being

(1) high performance
(2) ability to do the mission without a Earth capture aerobrake
(3) complete LTV reusability
(4) reusability it allows for the LEV (no aerobrake packing constraints as w the chemical sys)
(5) 90% propulsion system commonality with NTR Mars vehicle (tank sizing only difference) The ability to do all the SEI missions with an identical propulsion system looks attractive for several reasons

Two vehicles were developed; a single lander version, which does the same transfer mission as the present 90 day study chemical vehicle, and the double lander version which transfers two landers of any type per LTV sortie. The engine, shield, and truss system is purposely identical to the Boeing reference NERVA NTR vehicle to provide commonality. Possibly the best proof of concept and flight qualification methodology for a Mars vehicle propulsion system is to demonstrate it in space as a Lunar vehicle. A Lunar round trip is strikingly identical to a Martian round trip from a propulsion systems point of view if only all propulsive vehicles are considered. The primary difference being the difference in dV's and in propellant boiloff due to the longer Mars missions. For low energy Mars conjuction trajectories, the difference in dV from a Lunar mission is small.

Preliminary Lunar NTR Vehicles

designed to carry same payload as 90 day study reference chemical LTV/LEV system



IMLEO	single lander	double lander
T/W=3.5 (NERVA)	*197 t*	*293 t*
T/W=15 (PBR)	*170 t*	*267 t*

General vehicle characteristics:
- All propellant in single aft tank; no drop tanks
- Single 75000 lbf ref NERVA eng; wt=9684 kg, t/w=3.5, Isp=925 s
  Alternate engine: PBR; wt=2268 kg, t/w=15, Isp=925
- Shadow shield: 4500 (kg); identical to Mars NTR shield
- 45m SSF type truss (2400 kg) provides separation distance

Baseline single lander Lunar NTR vehicle:
- No aerobrake packing constraints; empty LEV returned to LEO for reuse. Allows checkout, refurbishment & resupply to be done in LEO vs LLO for current 90 day study chemical Lunar vehicle.
- IMLEO=196,560 kg;NTR t/w=3.5(*NERVA*);46.5 t LEV, 8.7t mod
  (=170,400 kg; NTR t/w=15 (*PBR*); P/L same as above)

Double lander version:
- Cargo lander mission done jointly with piloted lander mission
- Single NTR LTV does the job of two 90 day study chem LTV's
- Piloted LEV returned to LEO for reuse; cargo LEV left on surface
- Robust propulsion sys can accomodate wide range of payloads

Operational concerns:
- burn times:E dep:26 mins,L capt:6 mins,L dep:4 mins,E capt:9 m
  45 mins total. such short burn times keep fissionable product buildup small. Crew occupies the solar flare radiation shelter already required for the LTV mod during these short burns

- Both LTV and LEV are fully reusable vehicles

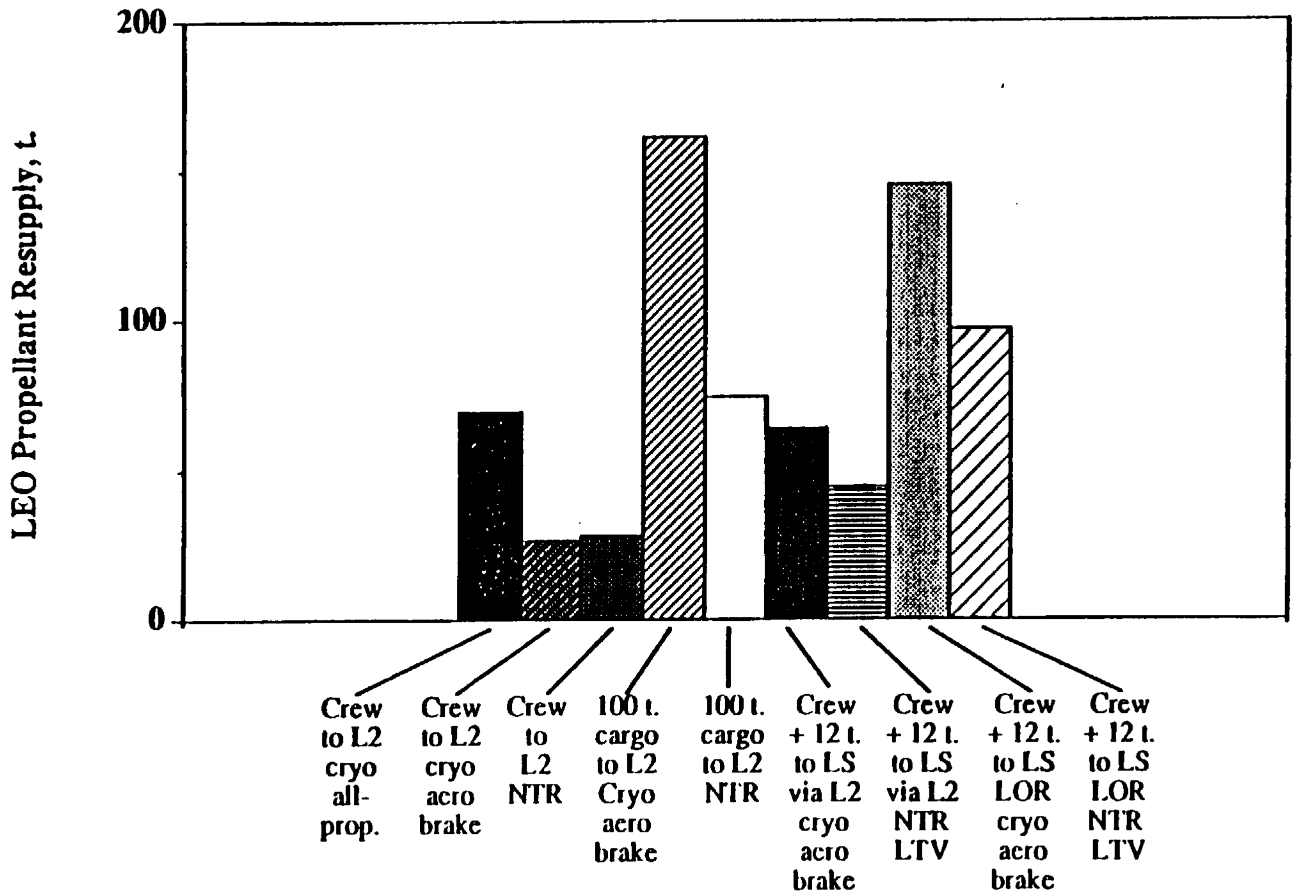
STCAEM/bbd/13June90

Cryogenic vs NTR LTV Comparison Preliminary Estimates

Scaling equations were used to compare cryogenic aerobraking and propulsive LTVs vs NTR propulsive LTV's. The first three bars on the left compare a crew mission to the L2 libration point. The payload is a 9 ton LTV crew module. The aerobrake was a 20% brake (20% of the total aerocapture mass). The advantage of aerobraking is indicated. In this case the NTR does not outperform cryogenic aerobraking because the mass of the 75k NTR engine and its shield overcomes the Isp advantage for this relatively low performance mission.

The next two bars compare heavy payload delivery to L2 and indicate a significant advantage for the NTR. The last four bars compare representative crew plus payload deliveries to the lunar surface: in the first case, L2 node using lunar oxygen, in the second case, LOR node not using lunar oxygen. In both cases the NTR performance is better than the cryogenic aerobrake. (Cryogenic aerobraking performance to the lunar surface for L2 and LOR nodes with lunar oxygen is equivalent. The difference here between LOR and L2 shows the advantage of lunar oxygen. If lunar oxygen is not used the LOR staging location is superior to L2).

Cryo vs. NTR LTV Comparisons
Preliminary Estimates



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Lunar NEP

This section contains a preliminary parametric analysis of Lunar NEP performance capabilities. The section contains the following

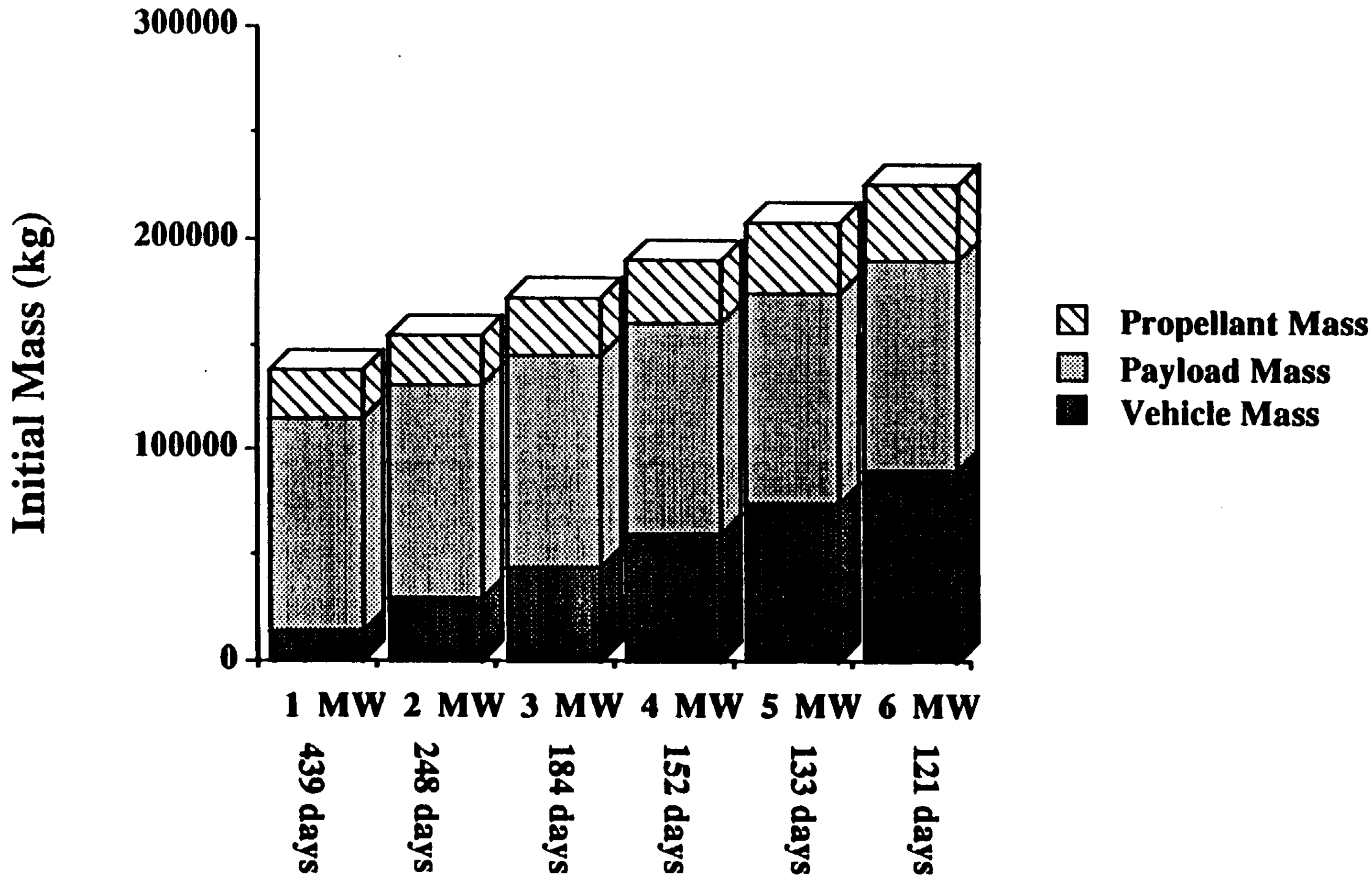
- IMLEO vs Isp for various power levels
- Propellant Mass vs Isp for various power levels
- Trip Time vs Isp for various power levels
- Payload Fraction vs Isp for various power levels

- Breakdown of IMLEO for various power levels - 10 kg/kW
- Breakdown of IMLEO for various power levels - 15 kg/kW

A power level of 3 MW (@ 5,000 sec Isp) will transfer 100 t of payload in less than 6 months. The IMLEO of this vehicle is ~150 t. The analysis assumes constant burn time and uses fundamental electric propulsion equations.

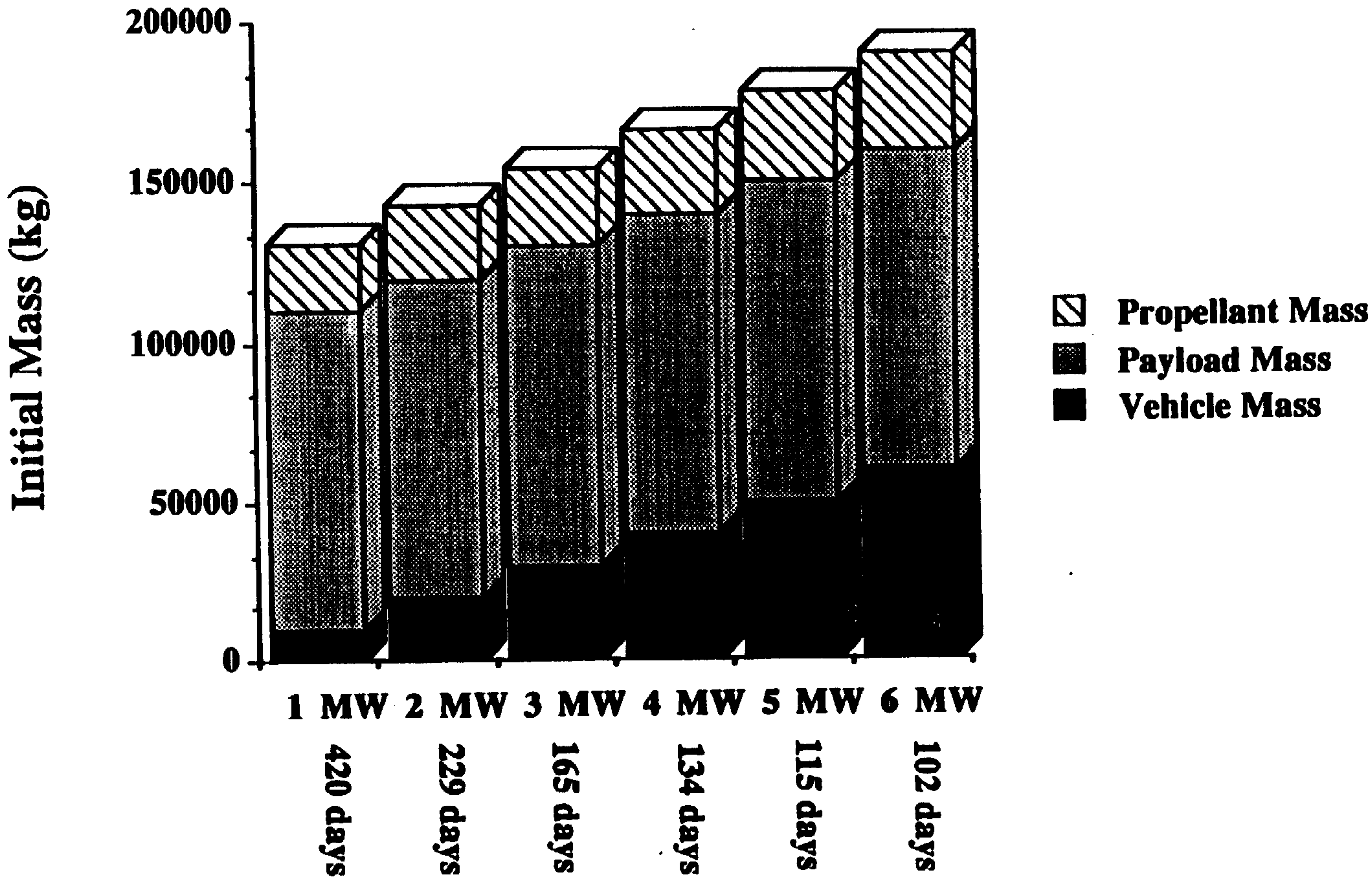
Lunar NEP Cargo Vehicle

ISP= 5000 sec, 15 kg/kW



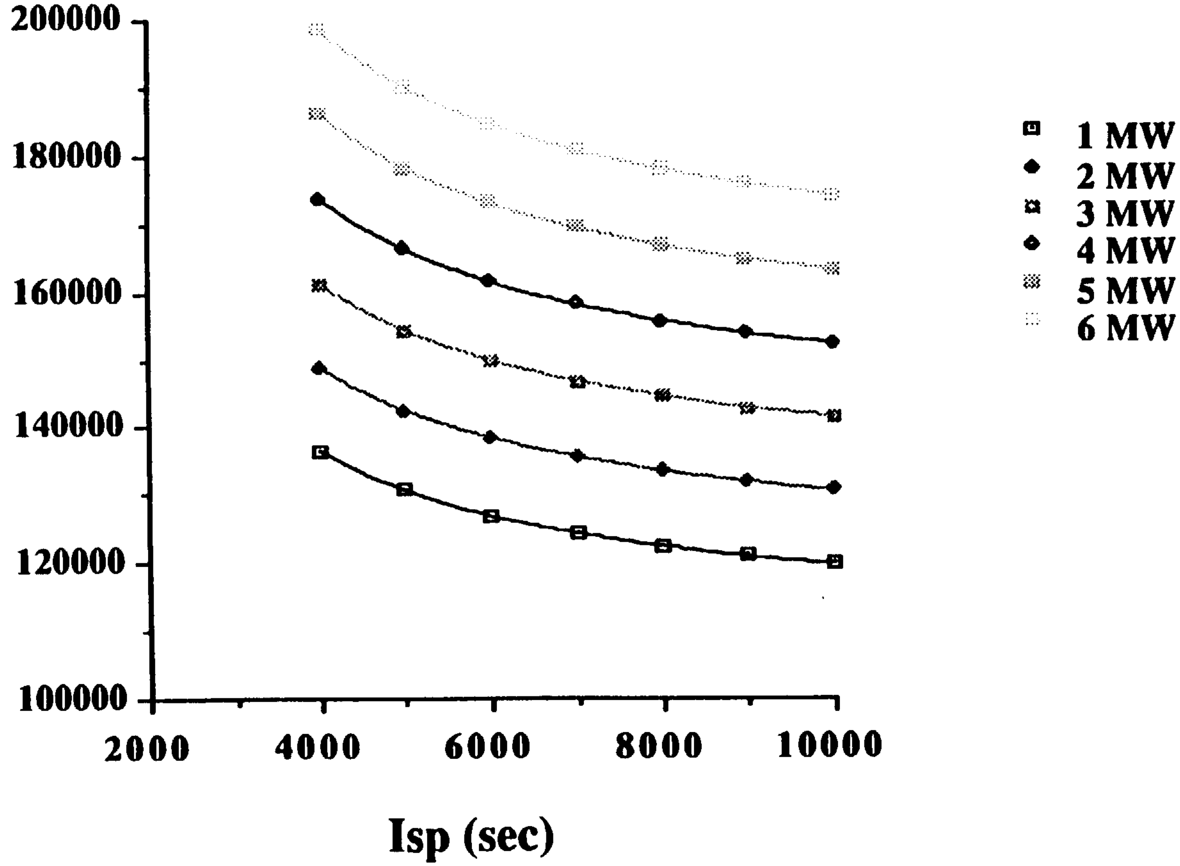
Lunar NEP Cargo Vehicle

ISP= 5000 sec, 10kg/kW

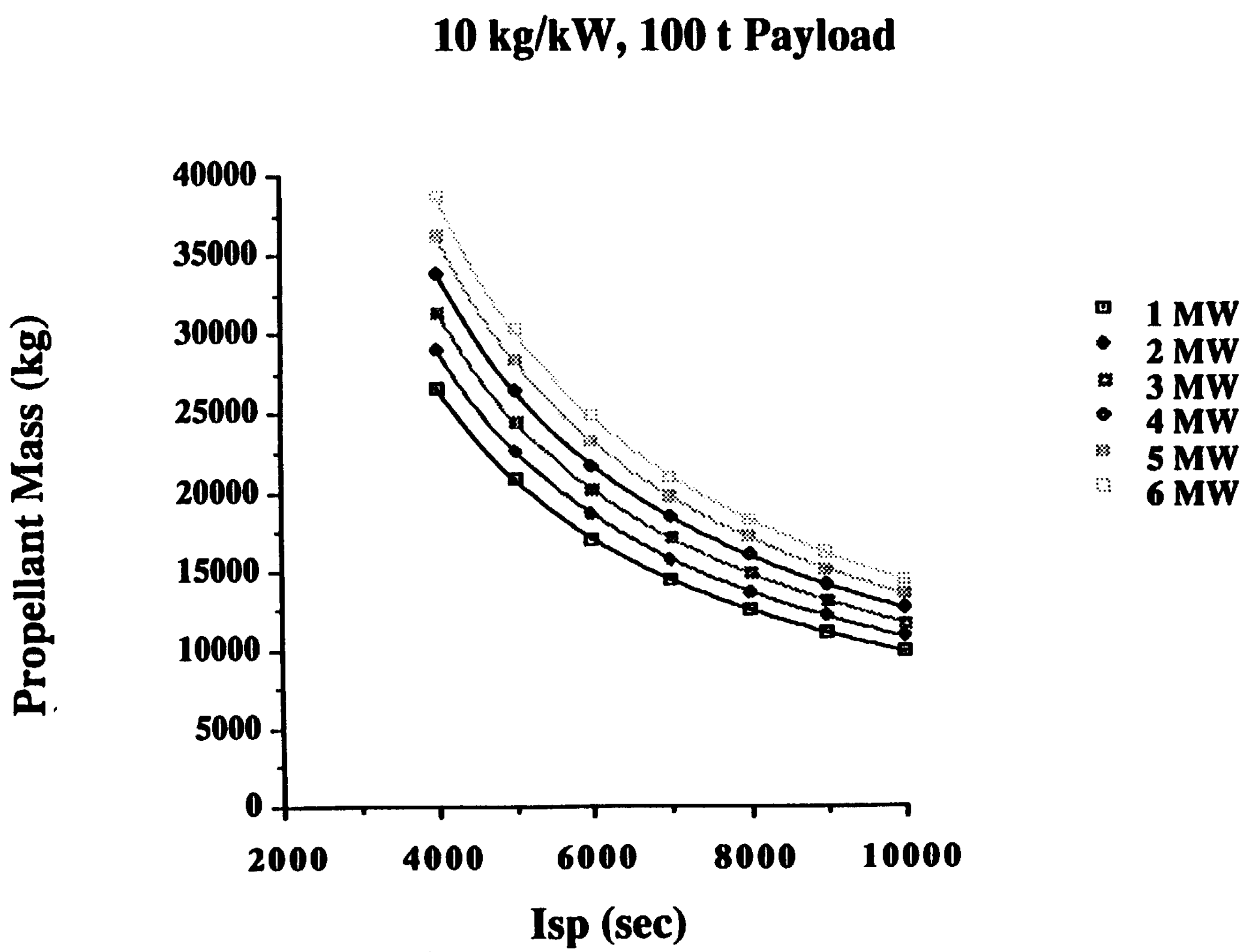


Lunar NEP Cargo Vehicle

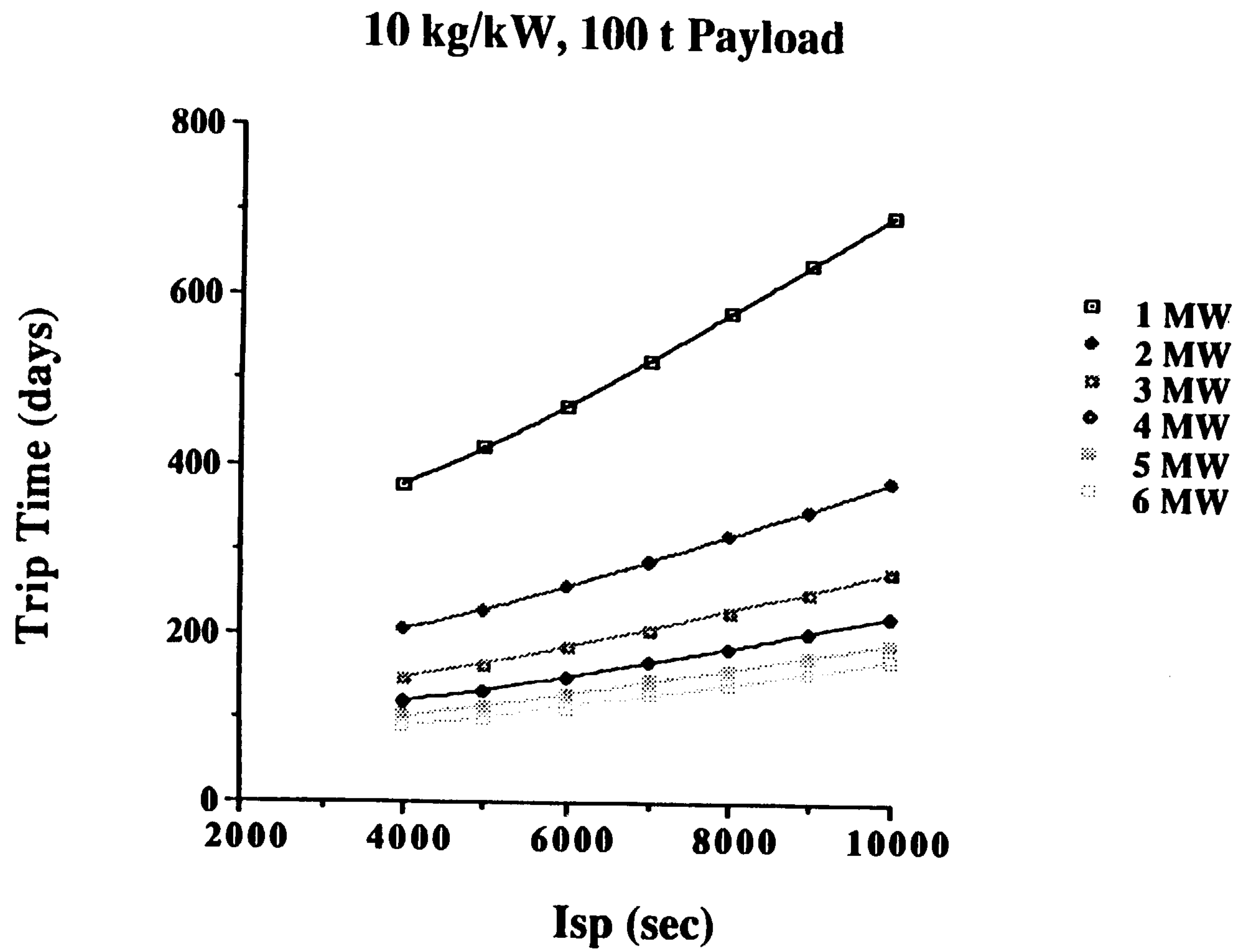
10 kg/kW, 100 t Payload



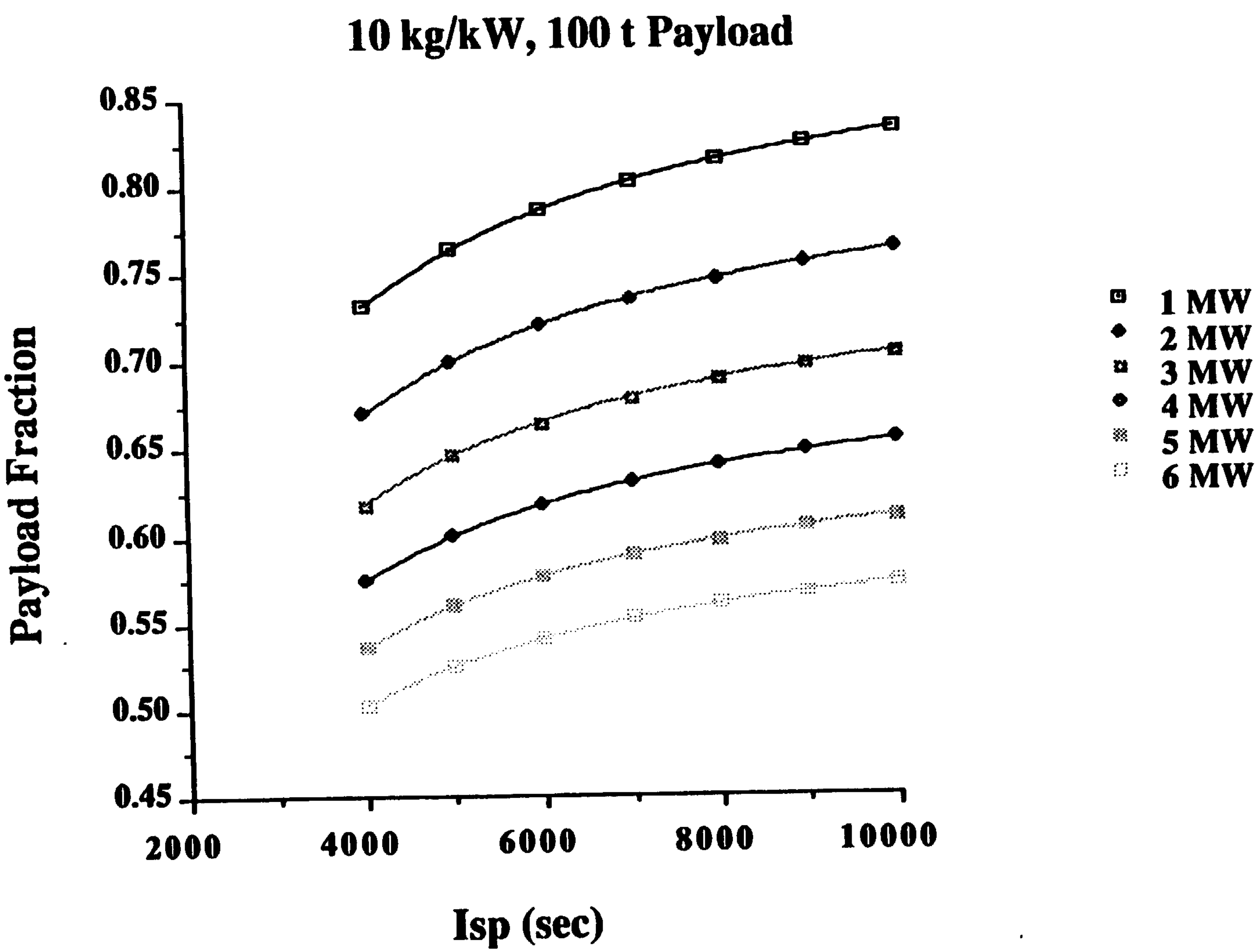
Lunar NEP Cargo Vehicle



Lunar NEP Cargo Vehicle

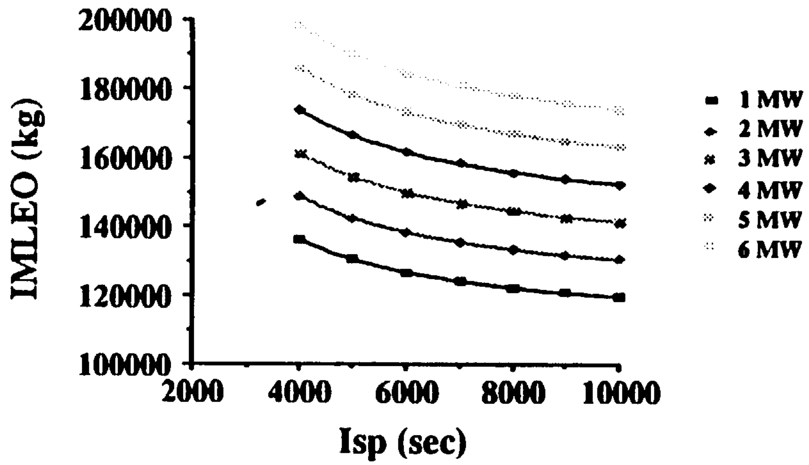


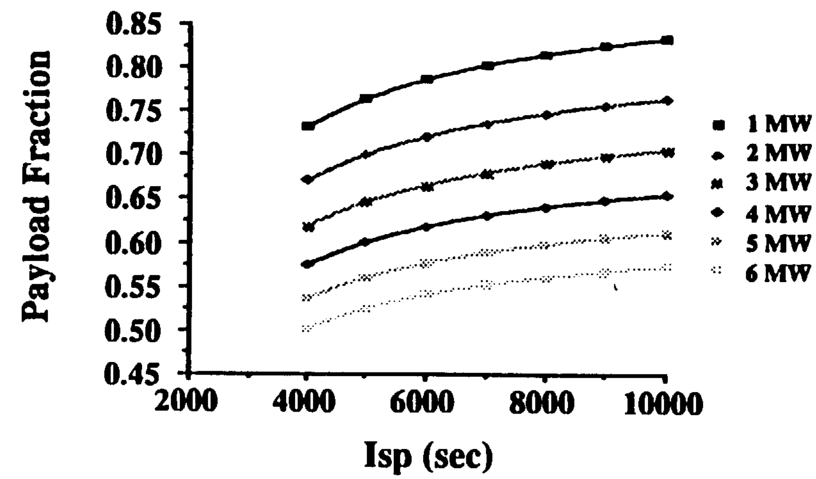
Lunar NEP Cargo Vehicle

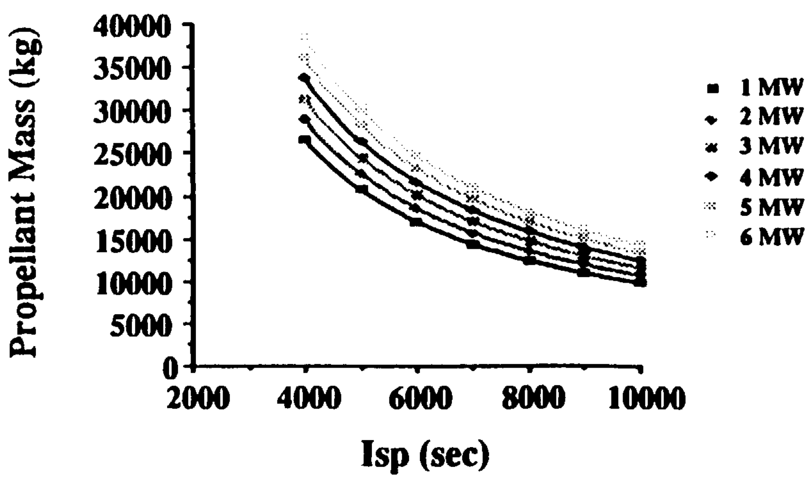


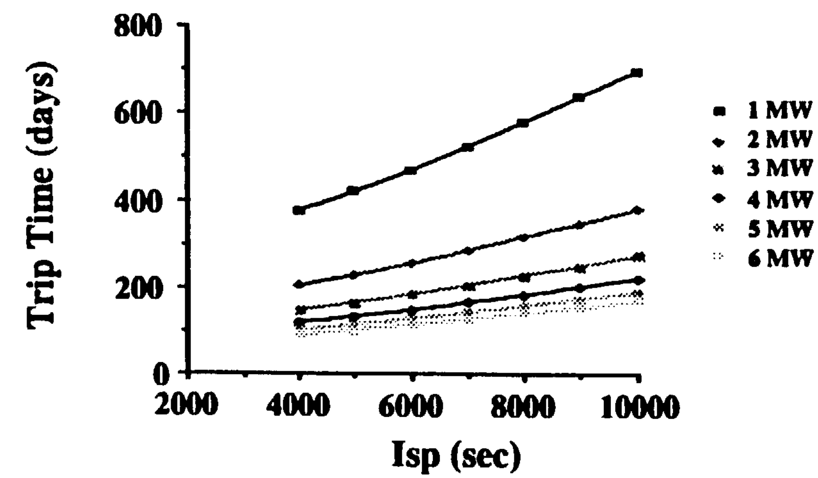
Lunar NEP Cargo Vehicle

10 kg/kW, 100 t Payload









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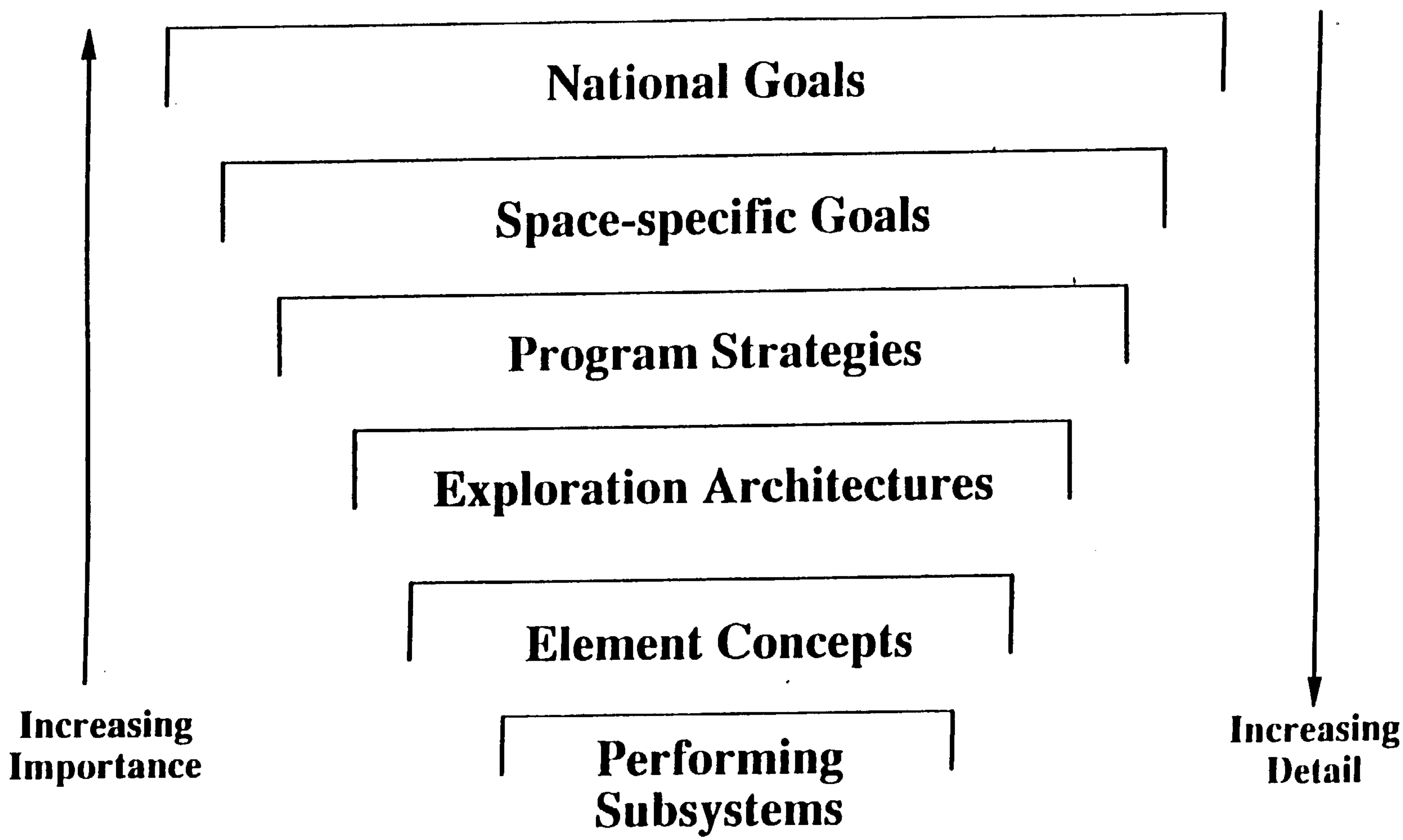
C. Architecture Matrix

Logical Types for Space Programs

Architectural planning for a space program deals with many levels of information. A major space program like the space exploration initiative must respond directivity to national goals in traceable ways. While we do not determine national goals, it is our business to understand how exploration architectures can be evaluated in terms of national goals.

National goals translate to space specific goals for specific exploration programs such as science emphasis or expanding human presence. These in turn can lead to program strategies for space-specific goals such as low risk, high technology, low cost and so forth. Finally, exploration architectures are integrated assemblages of systems, mission profiles, and operations, necessary to satisfy program goals.

"Logical Types" for a Space Program



Each logical type subsumes all the subordinate types

STCAEM/bs/11Jun90

Program Implementation Architectures

We have selected seven program implementation architectures for architectural analysis. These seven architectures incorporate the advanced propulsion options of principal interest in complete evolutionary architectural scenarios for lunar and Mars exploration. The facing page lists the features of each architecture and the rationale for selection of each.

Some of the architectures include suboptions. For example, the nuclear electric propulsion and solar electric propulsion architectures include optional use of the electric propulsion system for lunar cargo delivery from LEO to lunar orbit. The L2-based cryogenic aerobraking architecture includes use of NTR and NEP vehicles for LEO to L2 cargo delivery as options, and also includes a cryogenic all-propulsive conjunction mission option.

Program Implementation Architectures

Architecture	Features	Rationale
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	NASA 90-day study baseline
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.	High performance of nuclear electric propulsion
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.	High efficiency of solar electric propulsion; find cost crossover for array costs.
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.	High Isp of nuclear rocket enables avoidance of high-energy aerocapture at Mars.
L2 Based cryogenic/ aerobraking	L2-based operations; use of lunar oxygen.	L2 base gets out of LEO debris environment. Lunar oxygen reduces resupply by ~ factor 2.
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	Eliminates Mars orbit operations.
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	Eliminates boosting massive Mars transfer vehicle.

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SEI Program Scopes for Transportation Architecture Analysis

There are many space-specific goals and program strategies. We believe that transportation architectures will respond mainly to program scope. Some architectures are best suited to small program with early goals and others best suited to long range larger programs with ambitious goals. We have selected three representative scopes for small, moderate and large programs as illustrated on the facing page. These scopes permit definition of transportation requirements in terms of numbers of people and amounts of cargo transported to particular locations on particular schedules.

The second important feature of the scopes we intend to investigate is that they cover a scale factor greater than ten. A man tended science station may have a few people on the Moon for a short periods, or few people on Mars for short periods every other year. Permanent science bases will involve a dozen or so people. Industrial development of lunar resources on a scale of helium-3 scenarios leads to numbers of people presently estimated in the range of thousands by 2050. Beginnings of humans settlement of Mars involves numbers in the range hundreds to thousands. The 20-25 horizon for SEI is expected to permit growth in numbers of people only to dozens or so.

SEI Program Scopes for Transportation Architecture Analysis

Descriptor	Small	Moderate	Ambitious
Lunar Operations	Man-tended science station	Permanent science base 6 - 12 people	Industrial development of lunar resources
Mars Operations	Expeditionary visits ~4 people	Permanent science base 6 - 12 people	Beginnings of human settlement

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Earth-L2-Mars Mission Profile Schematic

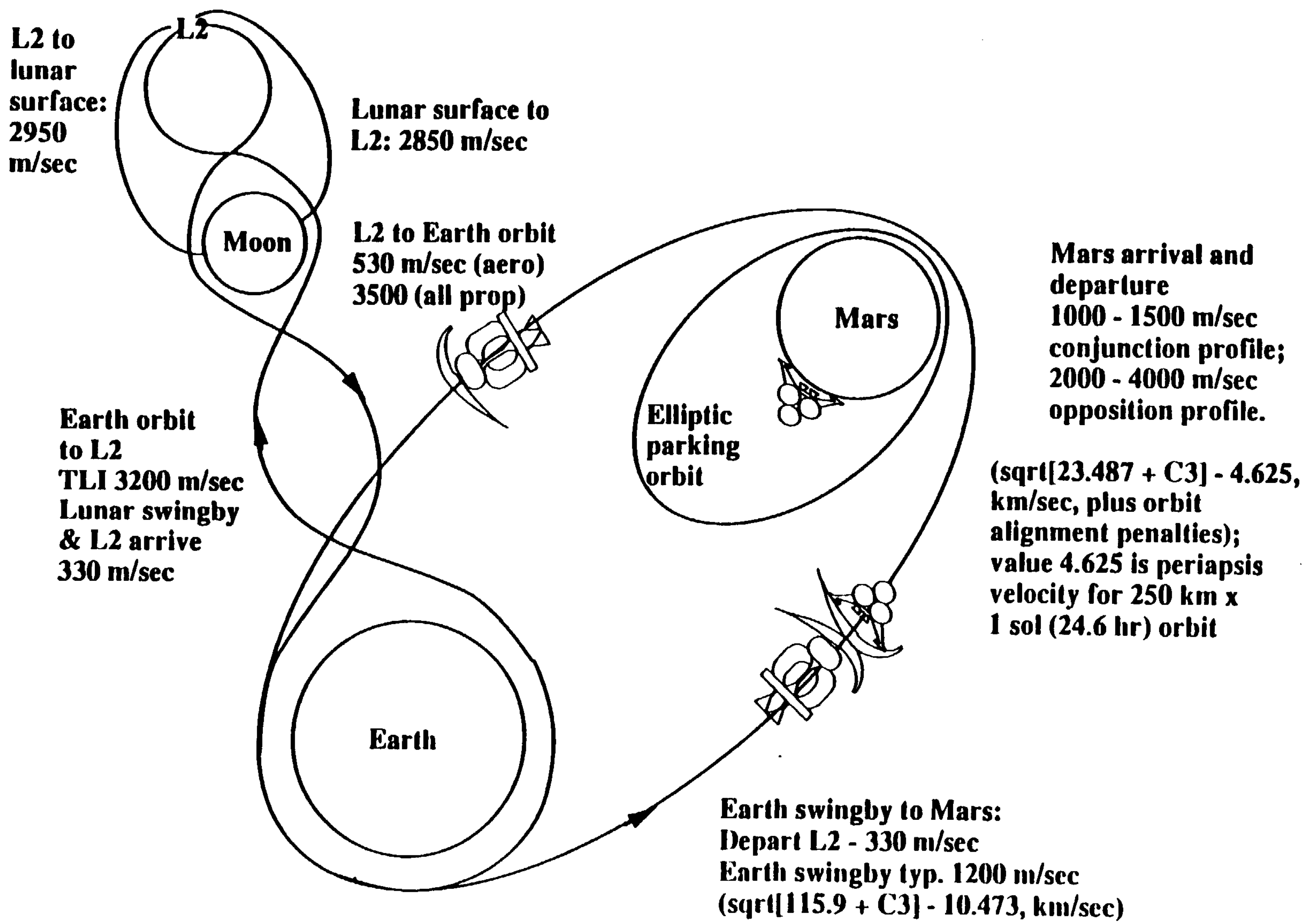
The use of the L2 libration point as a transportation node was originally suggested by the Farquhar and later by Keaton. The L2 node has three advantages. (1) the delta V from L2 to Mars transfer is almost 3000m/sec less than from low Earth orbit. As a result the vehicle is much smaller; (2) the transportation node is not in the low Earth orbit debris environment; (3) the L2 node is a suitable location for using lunar oxygen in Mars mission systems.

The L2 node scheme involves a network of transportation from the Earth to L2 and back, from L2 to the lunar surface and back, and L2 to Mars and back. The facing page indicates the delta V's for each leg of the transportation network.

Transportation to the surface of the moon via L2 is efficient when lunar oxygen is used for the LEV, as efficient as the low lunar orbit node with lunar oxygen. (The LEV becomes like the LTV in size). The disadvantage of the L2 node is that substantially greater lunar oxygen production is required compared to the low lunar orbit node. The advantage is that any site on the lunar surface is accessible any time and return to Earth is available at any time. Delivery of lunar oxygen to the L2 node by rocket is surprisingly efficient.

Transfer from L2 to Mars uses dual powered gravity assist, at the Moon and the Earth. This means that launches to Mars are limited to times the Moon is at proper location. The launch window problem is very similar to launching from a Earth orbit. In either case the launch window problem is difficult for opposition missions but for the much longer launch windows of conjunction missions, multiple chances are available. The return from Mars uses aero-assist to Earth and powered gravity assist from the Moon to L2.

Earth-L2-Mars Mission Profile Schematic



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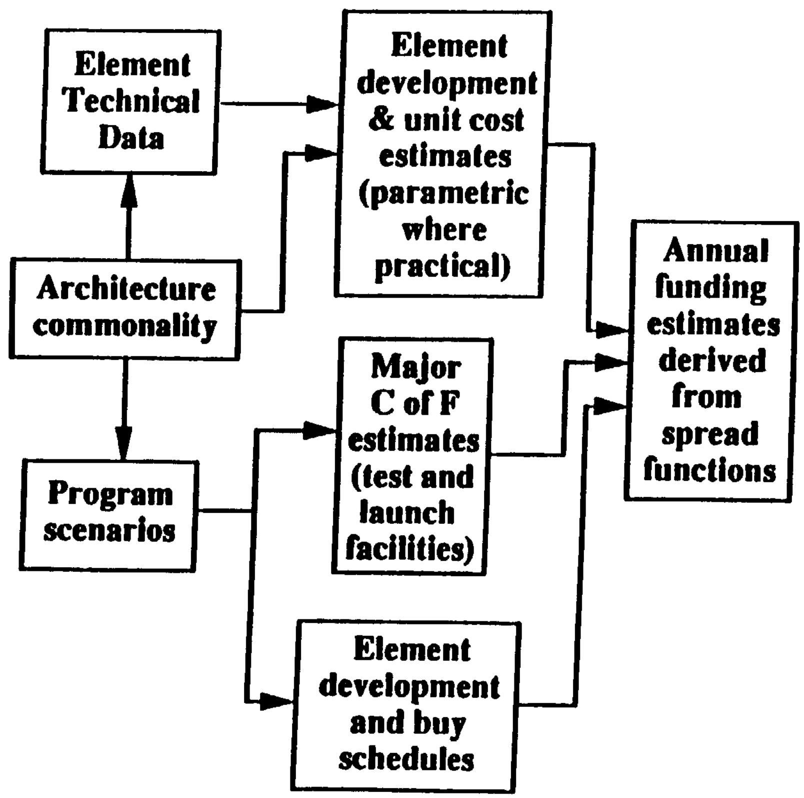
Life Cycle Cost Model Approach

Our basic cost model kernels are parametric cost models. We use the Boeing Parametric Cost Model and the RCA Price models to estimate development and unit cost. The determination of hardware to be costed comes from what architectural elements are needed and from element commonality of the architecture. Program schedules determine requirements and timing for major facilities and for the element development and buy schedules. All of these inputs are used to estimate annual funding for each component of the program, using cost spread functions. The costs are integrated into a spread sheet life cycle cost model to obtain annual funding for complete programs.

The ground rules used in this analysis are indicated on the chart.

The ground rule for use of closed ecological life support (CELSS) and lunar oxygen comes from economics trade studies conducted several years ago through last year.

Life Cycle Cost Model Approach



Ground Rules

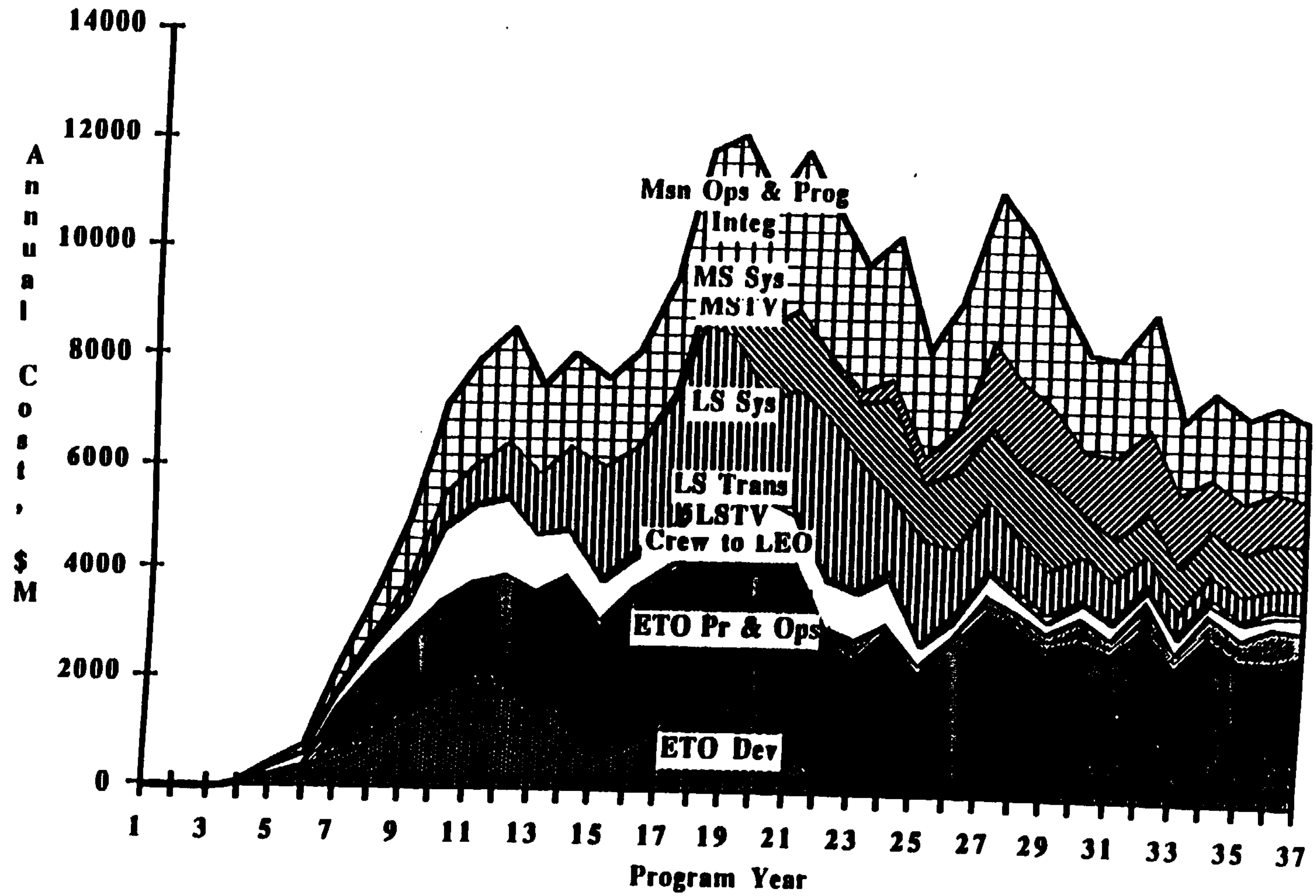
- No precursor missions costed.
- NASA contingency not added
- Common element in new application gets 25% delta DDT&E cost.
- No production learning unless production rate > 1 per year.
- Production rates maintained minimum of 1 per 5 years to keep lines open.
- Mission definitions flexible to enable transportation systems to operate at high efficiency.
- All scenarios include closed ecological life support and ISRU for efficiency.

VSTCABM/mha/31May90

Life Cycle Cost Example for Architecture Number 2 (NEP)

The facing page illustrates a typical output from our spread sheet life cycle cost model for architecture #2. The costs indicated are very preliminary; the chart serves mainly to indicate a rough range of cost for this architecture and nature of outputs expected. There are 3 points: (1) The Earth-to-orbit cost is larger than any of the others even though the Earth-to-orbit systems used in this architecture were modest in development and operations cost. This reinforces the view that advanced in-space transportation technology is a better investment than very large launch vehicles. (2) There is a large investment in lunar surface systems, even though they employ commonality with transportation and space station systems. This is caused by implementation of nuclear power systems for surface operations, and in part due to resource utilization equipment. If these cost estimates are correct, given the roughly 50% savings in launch cost, it appears ISRU is a wise investment. (3) Even though very advanced nuclear propulsion transfer system is used for Mars, the estimated investment for Mars surface systems is roughly equal to that for the Mars transfer systems.

LCC Example for Architecture #2 (NEP)



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Architecture/Launch Vehicle/Node Trends

Issues

- Launch vehicle size, shroud size, and lift capacity.
- Node complexity and cost.
- On-orbit assembly complexity
- Number of launches per year
- Development cost
- Per-mission cost

Trends from Architecture Analyses

- Large launch vehicle (up to 300 t. lift) does not eliminate on-orbit assembly.
- Keys to on-orbit assembly are (1) design the vehicle to keep it simple; (2) design for automation and robotics; (3) reusable space vehicles to reduce the frequency of assembly operations.
- Advanced in-space transportation technology reduces launch requirements enough that a 100-t., 10-meter shroud launch vehicle is adequate.
- Ultra-large launch vehicle results in high early program costs and is much more costly than advanced in-space transportation technology.
- Evolution and design for evolutionary transitions are the keys to affordable, efficient programs with long-term growth.

/STCABM/mha/31May90

Available Options

The facing page is a typical listing of the element options making up a total transportation architecture for SEI missions. The options listed are all candidates for incorporation into architectures. Trade studies have not eliminated any of these options. (The list is representative and not necessarily complete.) The number of options on this chart for each row of options is indicated on the far right. In most cases, any option can be combined with any other set of options. Thus, the total possible combinations number in the millions. It is clear that available future effort can not hope to examine all combinations. This drives us to a strategy for architecture sensitivities analysis, to develop key trends and conclusions from relatively few architecture combinations.

Available Options

								No. of options
ETO	100 t.	140 t.	200+ t.	Add prop tanker				3 x 2
Node	SSF	Separate	SSF + separate	Self-assy.	Wet tanks	Refuel vehicles	Propellant depot	4 x 3
Lunar mode	Direct	Direct/ lunar ox.	LOR	LOR/ lunar ox.	L2/lunar oxygen			5
LTV	Cryo all-prop	Cryo aerobrake	NTR	NEP/SEP cargo	Fully reusable	Partially reusable	Expend-able	4 x 3
LEV	Cryo	Storable	Combined with LTV	Fully reusable	Partially reusable	Expendable		3 x 3
Mars mode	2.7 year	1.5 year						2
MTV	Cryo all-prop	Cryo aerobrake	NTR	NEP	SEP	Cycler		6
Mars node	LEO	L2						2
MEV	Cryo	Storable	Combined with LEV					3

Total possible combinations 2,799,360

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Strategy for Architecture Synthesis

The strategy we have adopted is illustrated on the facing page. First, we examined propulsion systems options through trade studies to understand how they work and to define preferred configuration operating modes. Secondly, based on the knowledge gained through these trade studies we chose a set of architectures using combinations of systems and modes, paying attention to integration compatibility, evolutions and commonality. Third, we will compare and trade architectures over a range of scopes and obtain important sensitivities and understand how architectures respond to program scope. We expect this analysis to lead to preferred architectures for various scopes. The final step is to conduct trades within the winning architectures to make further improvements.

All of this is guided by knowledge of the architecture cost drivers described earlier and by the knowledge gained on how systems work together, from the trades conducted within individual propulsion systems.

Strategy for Architecture Synthesis

Conduct trades on systems to find preferred configurations and operating modes →	Synthesize candidate architectures using effective combinations of systems and modes →	Compare and trade architectures over a range of program scopes to obtain important trends and sensitivities →	Conduct trades within winning architectures to improve them
Criteria:	**Criteria:**	**Criteria:**	**Criteria:**
Risk Performance Cost Flexibility	Integration compatibility Evolution Commonality of hardware & technology	Ability to satisfy program goals & schedules Cost	Risk Performance Cost Flexibility

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Architectures Synthesis vs Mission/System Analysis

The facing page compares this approach to the traditional top-down systems engineering approach. The traditional approach shown on the right, starts with program goals, establishes mission requirements through trades, and continues to lower levels. As usually conducted, the traditional approach is faced with the great number of possible combinations noted earlier. The usual outcome is that requirement decisions are made and systems selected without trade studies.

The synthesis technique, on the left, attempts to avoid this problem by a combined top down/bottom up approach. It is similar to a classical optimization problem.

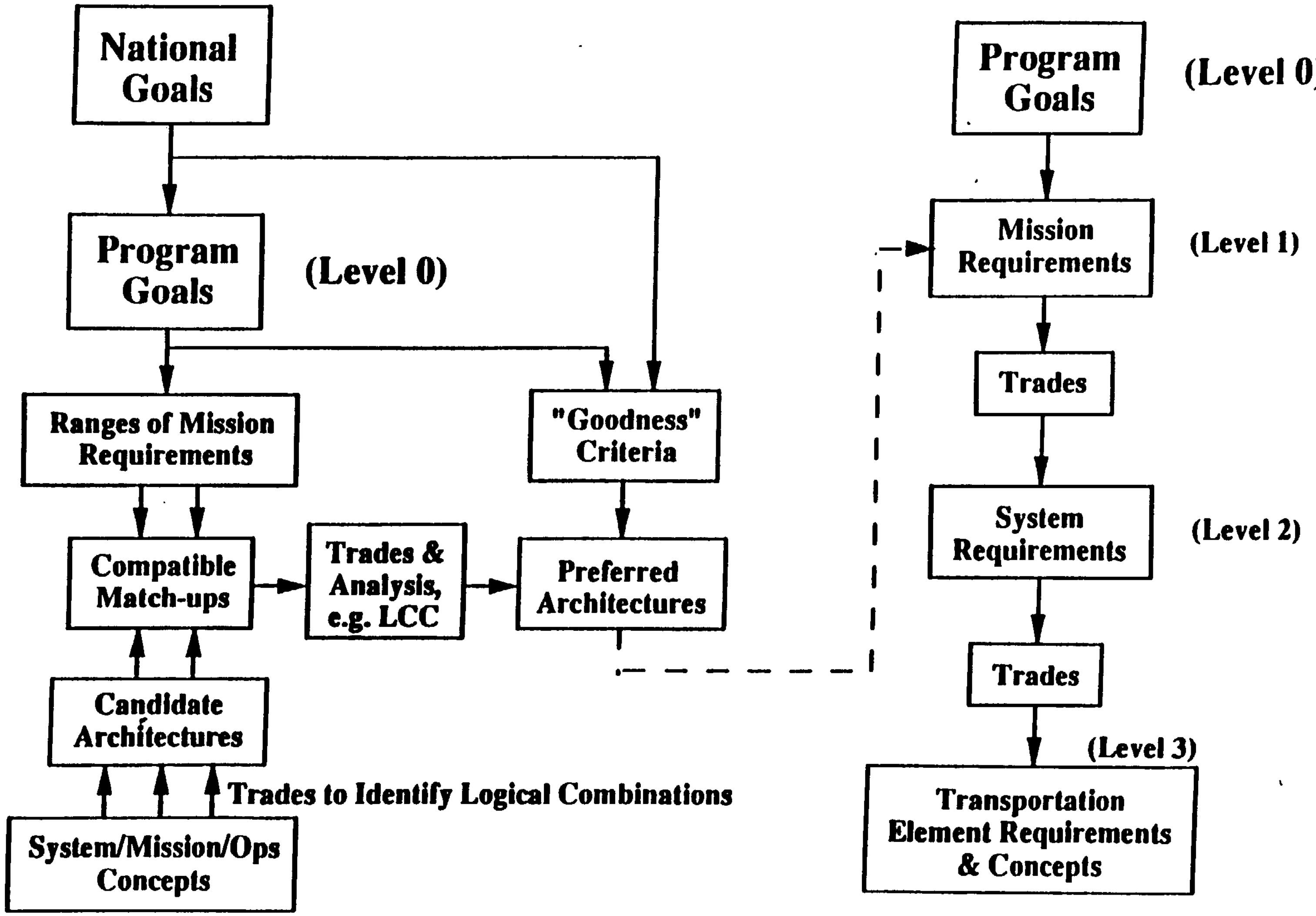
Optimization deals with infinite numbers of paths that satisfy boundary conditions. Optimization is a technique for generating only optimal paths. Any path that satisfies the boundary conditions is the sought optimal path.

Nothing quite as rigorous can be done in architecture synthesis. However, by bottom up trades, assembling systems into "good" candidate architectures, and matching with ranges of program scope, we may come close. The key is knowledge we obtain on what works well what things are compatible and combine well to satisfy mission requirements.

The last step is to conduct trades and analyses such as life cycle cost to identify preferred architectures, apply criteria derived from national goals program goals, to select among preferred architectures.

The dotted line indicates that one could then enter the traditional analysis flow with preferred architectures and their associated requirements and mission profiles, to further refine systems through systems engineering.

Architecture Synthesis vs. Mission/System Analysis



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Architecture Trade Flow

The facing page shows the low level system mission and operations trades that have been conducted or are being conducted for our seven architectures to represent the range of possible architectures for the SEI mission. Most of the trade areas have been presented in this briefing or have been presented in earlier briefings. The knowledge base in this area is fairly complete except that only very preliminary analyses have been done for the cryogenic direct mode and for cycler orbits. When these two options are completed we will be ready to finish up the architecture analysis.

Architecture Trade Flow

Cryo/Aero-braking	Nuclear Electric (NEP)	Solar Electric (SEP)	Nuclear Thermal Rocket (NTR)	L2/Lunar Oxygen	Cryo/Aero braking Direct	Cycler Orbits

• Mission design
• Reuse
• Aerobrake
  • shape
  • heating
  • GN&C
  • structures
  • assembly
• All-propulsive conj. option
• Modularity & commonality
	• Mission design
	  • trip time
	  • gravity assist
	  • node location
	• Power cycle
	• Power level
	• Specific power
	• Redundancy mgmt.
		• Mission design
		  • trip time
		  • gravity assist
		  • node location
		• Solar cell type
		• Power level
		• Specific power
		• Assembly/ deployment of large space structure
			• Mission design
			• Isp and T/W sensitivity
			• Reuse
			  • tanks
			  • engines
			  • core stage
				• All-propulsive conj. option
				• Lunar oxygen benefits
				• Integration of lunar & Mars ops.
				• Advanced propulsion for LEO-L2 operations
					• Performance vs. separate MTV/ MEV
					• Sensitivity to propellant choice
						• Mission design
						• Feasibility of high Mars encounter velocities
						• Design of "taxis"
						• Operational integration

For all: Overall configuration; key subsystems performance; integration compatibility; operations analyses

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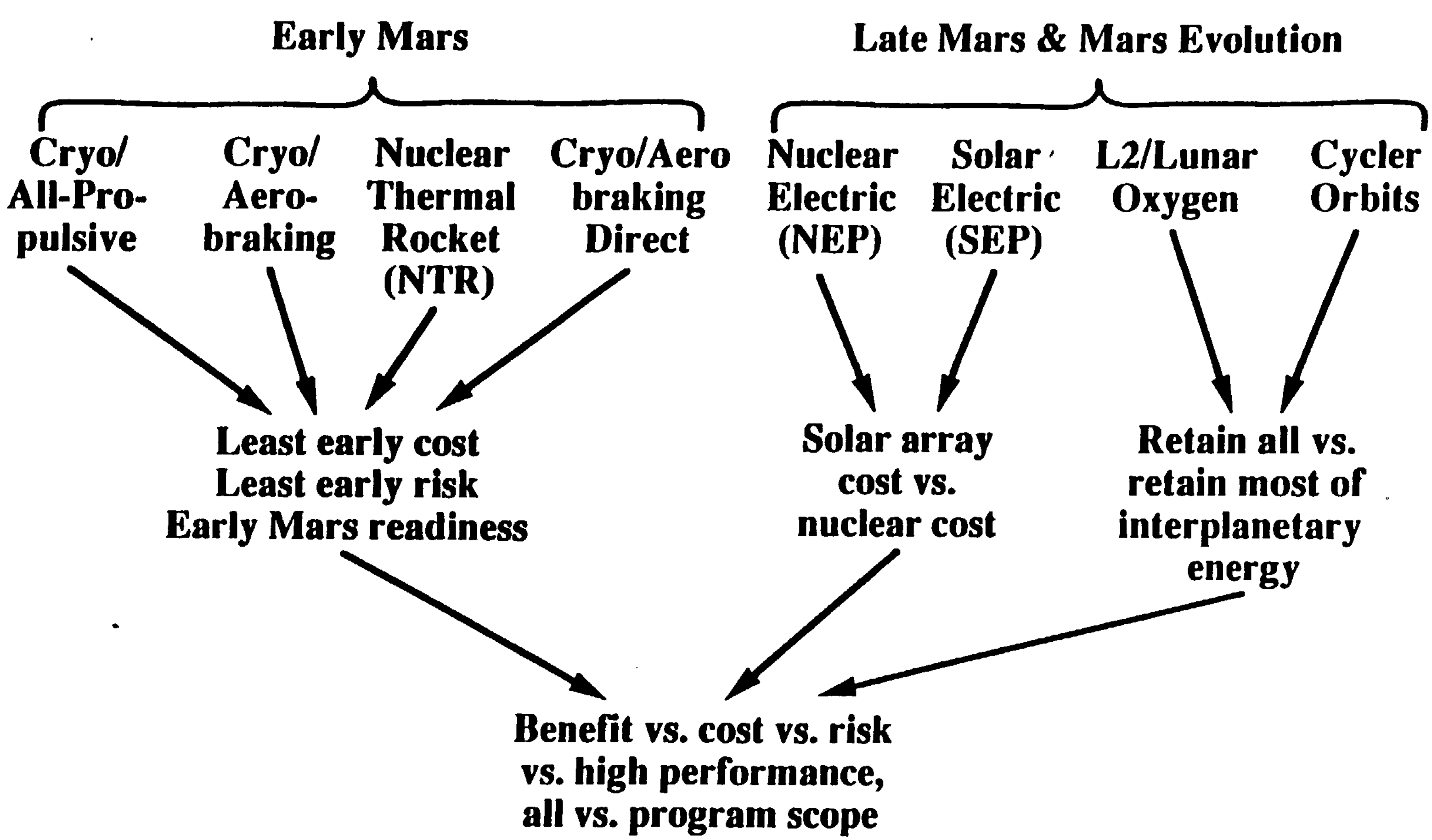
Architecture Evaluation Approach



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Architectural Cost Drivers

Our investigations of architectures, while preliminary, indicate the importance of cost drivers, in the order listed on the chart. The number of development projects should be minimized through commonality and phased by evolution so that development costs are reduced and are spread over the life cycle of the program, rather than lumped early in the program.

Space hardware for SEI missions is expensive and should be reused it if possible. As an example, our unit cost estimate for the Mars transfer crew module is more than a billion dollars. Reuse of this equipment motivates investment in the advanced transportation technology needed to make it reusable.

The third point is that Earth launch mass drives Earth launch cost. Even if Earth launch cost is reduced by ALS-class vehicles, the Earth launch cost is the largest single part of program cost.

The final point is that design and development of systems with mission and operation flexibility enhances commonality and minimizes the risk that changes in mission requirements force new developments or major changes.

Architecture Cost Drivers

- Number of development projects (minimize through commonality)
- System reuse (maximize)
- Earth launch mass (minimize)
- Mission and operational flexibility (maximize)

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Program Implementation Architectures Relation to Aerobraking

The facing page indicates uses of aerobraking for the various architectures. As noted, some form of aerobraking occurs in all of the architectures, in particular for Mars landing and for Earth capture on return from lunar missions. In addition, some of the architectures include an Earth crew capture vehicle (ECCV) for direct return of the crew to Earth in cases where, for example, an NEP or SEP vehicle must spiral back down to LEO or in the case of an NTR where the vehicle captures into a highly elliptic orbit.

Program Implementation Architectures

Architecture	Features	Aerobraking Functio Mars cap	 Mars land	 Earth cap/ lunar	 Earth cap/ Mars	 Earth entry*
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	x	x	x	x	x
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.		x	x		x
L2 Based cryogenic/ aerobraking	L2-based operations; optional use of lunar oxygen.	**	x	x	x	x
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	x	x	x	x	
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	***	x	x	x	x

Notes: * optional/emergency mode **opposition class only *** MEV-class crew taxi (not a large MTV)

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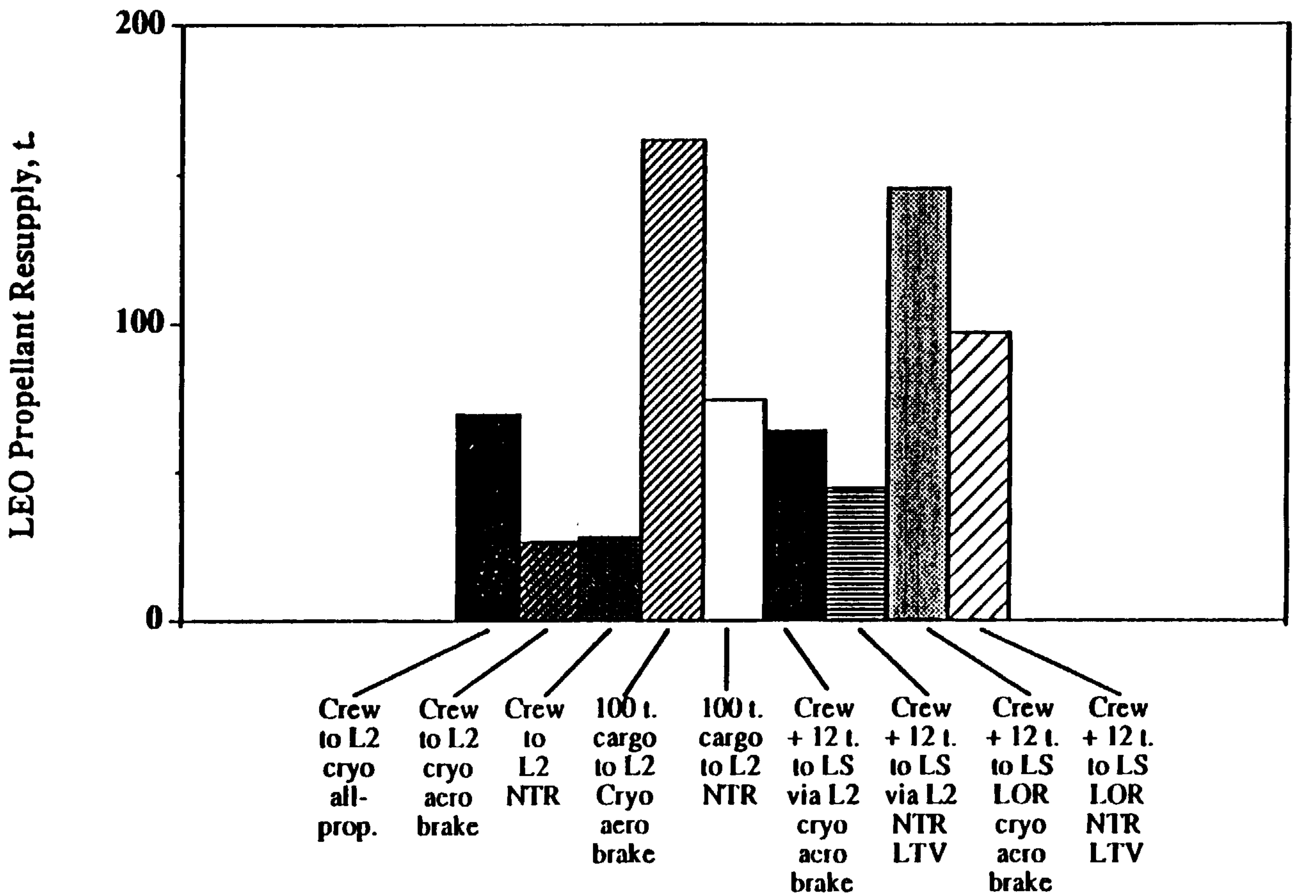
Cryogenic vs NTR LTV Comparison Preliminary Estimates

Scaling equations were used to compare cryogenic aerobraking and propulsive LTVs vs NTR propulsive LTV's. The first three bars on the left compare a crew mission to the L2 libration point. The payload is a 9 ton LTV crew module. The aerobrake was a 20% brake (20% of the total aerocapture mass). The advantage of aerobraking is indicated. In this case the NTR does not outperform cryogenic aerobraking because the mass of the 75k NTR engine and its shield overcomes the Isp advantage for this relatively low performance mission.

The next two bars compare heavy payload delivery to L2 and indicate a significant advantage for the NTR. The last four bars compare representative crew plus payload deliveries to the lunar surface: in the first case, L2 node using lunar oxygen, in the second case, LOR node not using lunar oxygen. In both cases the NTR performance is better than the cryogenic aerobrake. (Cryogenic aerobraking performance to the lunar surface for L2 and LOR nodes with lunar oxygen is equivalent. The difference here between LOR and L2 shows the advantage of lunar oxygen. If lunar oxygen is not used the LOR staging location is superior to L2).

Cryo vs. NTR LTV Comparisons
Preliminary Estimates



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Lunar Oxygen Delivery Mission Description

	Delta V m/sec	Mass Ratio	Propellant Used	Mass Remaining	
				71,729 kg.	(IMLEO) Resupply = 61,742
TLI	3110	1.95	34,938	36,791	
Lunar pass & L2 arrive	330	1.073	2,516	34,275	LTV stage 7,902; Aerobrake 2,085; Hydrogen for LEV 23,085; Return propellant 1,203
				76,851	(LTV start descent)
Descent from L2	2950	1.884	36,056	40,796	(Landed)
				274,298	(Liftoff)
Ascent to L2	2850	1.844	125,530	148,767	LEV Stage 16,639; Landing legs 1,224; Descent LO2 30,904; LO2 net to L2 95,000; LO2 tank 5,000
L2 depart & lunar pass	330	1.073	765	10,426	
Aerobrake to LEO	200	1.044	438	9,987	LTV stage 7,902; Aerobrake 2,085

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Lunar Summary

- Seven beneficial mission modes, all captured by LTV-like and LEV-like vehicles.
- Tailoring LTV and LTV to one or two modes loses flexibility.
- ISRU (lunar oxygen) beneficial to lunar and Mars transportation. Needs end-to-end economics trade.
- Early expendable mode with ECCV (modified ACRV?) can bypass Space Station Freedom; LTV & LEV evolve to reuse.

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II. Requirements, Guidelines and Assumptions

A. Level I, II and III Requirements

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Lunar IP&ED Text
[*Editorial note: viewfoil charts are referred to as 'charts'. If the convention in the IP&ED documents is to refer to them as 'figures' or 'tables',do a find and replace operation (⌘H in Microsoft Word 4.0)*]

II. Requirements and Assumptions

II.A. Levied Requirements

There is not a controlled baseline (system specification and configuration) for this study. This section includes our best understanding of what NASA would define as requirements if they had to be baselined at this time. It also includes lower level requirements and assumptions we derived or made respectively.

Level I requirements (first chart) are concerned with overall program schedule, mission, funding, and interface to other programs. Level II requirements (charts 2 through 4) are grouped by systems: Earth-to-orbit (ETO) transportation, ETO support facilities, space transportation vehicles, crew transfer module, and Lunar surface system interface.

The level I requirement of a first cargo landing in 2000 lead to level II requirements for ETO transportation test flight and Space Station Freedom (SSF) support readiness in 1999. The Level I requirement to use SSF leads to specific SSF accommodations requirements in the Level II requirements.

Level III requirements include vehicle specific requirements. The fifth chart lists requirements levied on the space transportation vehicles. These are propulsion characteristics: cryogenic propellant with an aerobrake; design margins; and operational characteristics: boiloff of cryogens, propellant transfer capability, and baseline orbit.

II.B. Derived Requirements

In the course of the study, derived requirements were developed from the levied requirements and the analyses. In some cases changes to the levied requirements are recommended. Issues addressed by the derived requirements in the mission area include the reference mission, basing, on-orbit assembly, modularity, and life (chart six). Other areas addressed in derived requirements are habitation module, propulsion, and transfer vehicle requirements (chart seven), and excursion vehicle and aerobrake requirements (chart eight). Excursion vehicle requirements include performance requirements and design requirements.

II.C. Assumptions

Assumptions have to be made in the initial cycle of analysis because all the data required is not yet available. In later analysis cycles, these initial assumptions will either be validated or replaced by more correct values. The subjects for which assumptions have been recorded are crew size, cargo capacity, aerobrake characteristics, ETO vehicle capacity, mission mode, and engine out capability.

III. Mission Operation

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Lunar Transportation Family
Level I Requirements

- Lunar cargo landing in 2000
- First manned landing in 2001
- Manned occupation of the lunar surface:
  - 4 crew for 30 days in 2001
  - 4 crew for 6 months to 1 year in 2004
  - 12 crew for 2 years in 2012
- Space Station *Freedom* (SSF) serves as operations base in the 1990's --- Modified to perform lunar transportation node functions in 2009
- 2 lunar flights per year rate (objective) --- mass to lunar surface TBD
- Lunar LOX production capability in 2009
- Minimize transportation impact on science at SSF (goal)
- Steady annual investment
- Cost analysis covers:
  - Next 5 years (1991-1996)
  - Funding through 2001
  - Sensitivity of long-term cost to schedule flexibility and potential technology advancements

Lunar Transportation Family

Level II Requirements

Earth-to-Orbit Transportation

- Test flight in 1999
- STS to be used for crew launches and selected cargo to SSF
- Shuttle-C BLD II and ALS are launch vehicle alternatives for delivering cargo to SSF orbit
  - First test flight in 1998
  - Not more than 6 flights/year (Shuttle-C)
  - TBD dia. payload capability
  - TBD performance capability
- All launches out of ETR
- ETO consists of pre-launch activities through SSF rendezvous
- Lunar crew and cargo returns to Earth via scheduled STS resupply missions

Earth-to-Orbit Support Facilities

- SSF will be capable of supporting the lunar program in 1999
- Minimal vehicle assembly activities in LEO desireable
- SSF is fully operational node to support transportation in 2000
- SSF accommodates 1 LTV/LEV/Aerobrake/Cargo until 2004, thereafter 2

Lunar Transportation Family
Level II Requirements

Space Transportation Vehicles

- SSF and LSS provides meteoroid/debris protection for transportation
- Chemical propulsion (LO2/LH2)
- Flight test of LTV/LEV in 1999
- 33 t delivered to the surface in a cargo mode
- 4 crew + cargo to the surface in crew mode
- LTV/LEV independent of lunar LOX availability assumptions
- Departure from Earth orbit with lunar orbit rendezvous (no direct landing)
- In-space refueling capability
- Autonomous rendezvous and docking
- Minimum vehicle activities required in LEO
- LTV/Aerobrake maintenance at SSF --- LEV maintenance at LS
- Vehicle reusability:
  - Initiated in 2004 for LTV
  - Initiated in 2009 for LEV (after lunar LOX production begins
  - 10 mission reuses
  - No major refurbishment in space/lunar surface (goal)
  - Demonstrated before manned aerobrake mission
- LEV lands on 50 m dia. lunar pad
- Technology to level 6 by 1993

Lunar Transportation Family
Level II Requirements

Crew Transfer Module

- Supports crews of 4, 6, and 8 --- Designed nominally for 4
- Surface stay times begin with 30 day --- stretch to 2 years
- Separate modules for LEO to LLO and LLO to surface and surface rover

Lunar Surface System (LSS) Interface

- Surface systems remove/load payload --- unloads within 48 hours of landing
- Transportation maintains safe/healthy payload from LEO to LS
- LSS provides airlock for crew egress from LEV
- LSS provides remote payload control capability after landing
- Propellant offload capability provided by LSS for stay times > 1 lunar day
- LSS provides navigation aids for LS pad landing
  - 50 m dia.
  - < 2° of slope
  - Irregularities < 0.2 m
- Solar flare protection provided by LSS

Lunar Transportation Family

Level III Requirements

Space Transportation Vehicles

- LO2/LH2 propellant
- Aerobrake
- Boiloff of cryogenic propellant
- Design contingency on dry mass:
  - Existing hardware = 5%
  - Modified hardware (existing/moderate tech.) = 10%
  - New hardware (advanced tech.) = 20%
  - Performance flight reserves = 2%
- Propellant transfer capability
- Baseline lunar orbit = 300 km

B. Derived Requirements

Lunar Transportation Family

Derived Requirements

Mission Level

- Vehicles to accommodate the following missions:
  - Tandem Direct initially
  - Evolve to Lunar Orbit Rendezvous (LOR)
  - L2 missions accommodated with Tandem Direct vehicles
  - Stand alone "sortie" mission to a site of interest
    - 4 crew
    - Daytime only (14 days)
- Space based vehicles
- On-orbit assembly required, however minimized
- Vehicles designed to be integrated, intact, into a 10 m x 30 m by 100 t launch vehicle
- Vehicle to consist of a "kit-of-parts" that can be reconfigured to accommodate evolving mission needs
- Vehicles designed to maximize vehicle reusability
- Minimum of 5 reuses with up to 30 year servicing life
- System evolution to Mars rather than entire vehicle commonality
- System flexibility for changing mission modes:
  - Changing crew size
  - Changing payloads
  - Increasing propellant loads

Lunar Transportation Family
Derived Requirements

Habitation Modules

- All habitation modules sized volumetically from historical spacecraft volumetric data
- Direct visibility of all landing and docking procedures
- Size hab modules for crew of 6 for 7 days for evolution scenarios

Propulsion System

- Maximum engine gimball = 18.5° to limit steering loss to 5%
- 1 engine-out capability
- Advanced RL-10 30 klbf. engines
- Throttling or gimballing capabilities for steering in engine-out mode

Transfer Vehicle

- Capability to rendezvous with Space Station *Freedom*
- Accommodate 4 crew initially for up to 28 days
- Single up/down orientation during both landing and aerobraking
- Contiguous transfer to Excursion Vehicle

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Lunar Transportation Family

Derived Requirements

Excursion Vehicle

- Return crew of 4 and 500 kg of samples to LLO
- Deliver 33 t of cargo to LS in non-reusable mode
- Deliver crew of 4 + nominal payload in crew reusable mode
- Designed to accommodate varying, nonsymmetrical payloads
- Payload integration optimized for surface access
- Descent abort desireable through payload jettison
- Crew module placement optimized for surface access and visiblity
- Contiguous transfer from LTV
- Sustain 4 crew for 3 to 5 days
- An airlock is mandatory for lunar surface residences beyond 3 days
- Landing gear requirements:
  - Designed for 30° slope
  - Max. surface undulation between footpads = 1 m

Aerobrake

- L/D = 0.25
- Sized to accommodate a reusable system
- Derivative of the Mars L/D = 0.5 shape

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C. Assumptions

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Lunar Transportation Family
Assumptions

- Crew of 4
  - size the hab modules for 6 crew for growth to medium and large scale missions
- Payload to the surface
  - 33 t in cargo expendable mode
  - 4 crew + payload in crew reusable mode
- Aerobrake
  - L/D = 0.25
  - Shape derived from Mars L/D = 0.5 shape
- Launch vehicle
  - 10 m x 30 m shroud
  - 100 t to LEO
- Mission mode
  - Tandem direct initially
  - Evolve to LOR
  - Accommodate evolving mission goals
- Single "engine-out" capability
  - T/W = 1.6 for lunar descent
  - T/W = 0.4 in transit
  - RL-10 derivative, 30 klbf engines

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III. Mission Operations

A. Mission Analysis and Performance Parametrics

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III. Mission Operations

The mission operations section includes data on mission analysis studies and performance parametrics as well as the operating modes and performance evaluations which include the STCAEM recommendations.

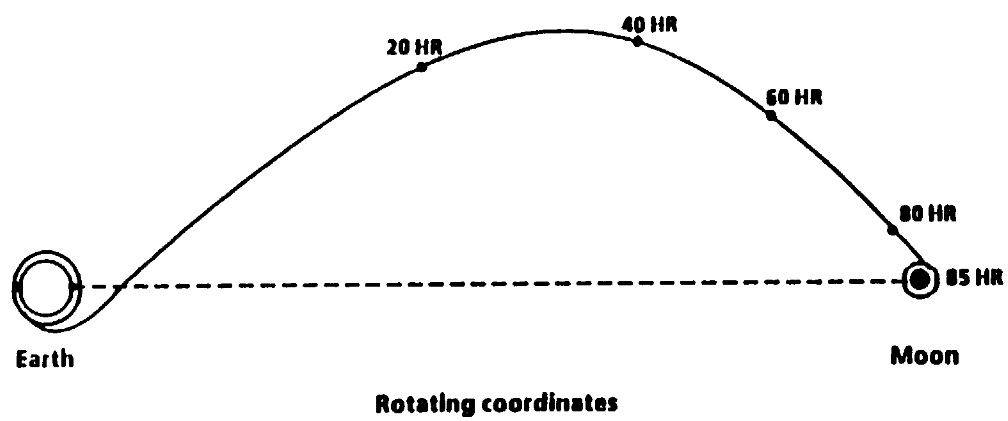
A. Most of the lunar **mission analysis and performance data** was generated during the 90-day study time-frame. Included in this document is data on timing of translunar trajectories, the lunar orbit insertion ΔV for different arrival asymptotes, transit time, and opportunity for lunar transit leaving from the vicinity of Space Station *Freedom.*

B. Initially, we identified seven **lunar vehicle modes** which could be implemented with a pair of vehicles; an LTV-like and an LEV like vehicle. During the course of configuring, sizing and generating performance data on these vehicles, 11 mission modes were identified which can be implemented with 5 major elements; a 110 t propulsion stage, a 25 t propulsion stage, a transfer hab, a crew cab and an aerobrake. Implementation of the various mission modes is accomplished by "piecing" together the required elements. The result is a Lunar Transportation Family (LTF) that is flexible and can evolve to meet growing mission needs and changing mission modes.

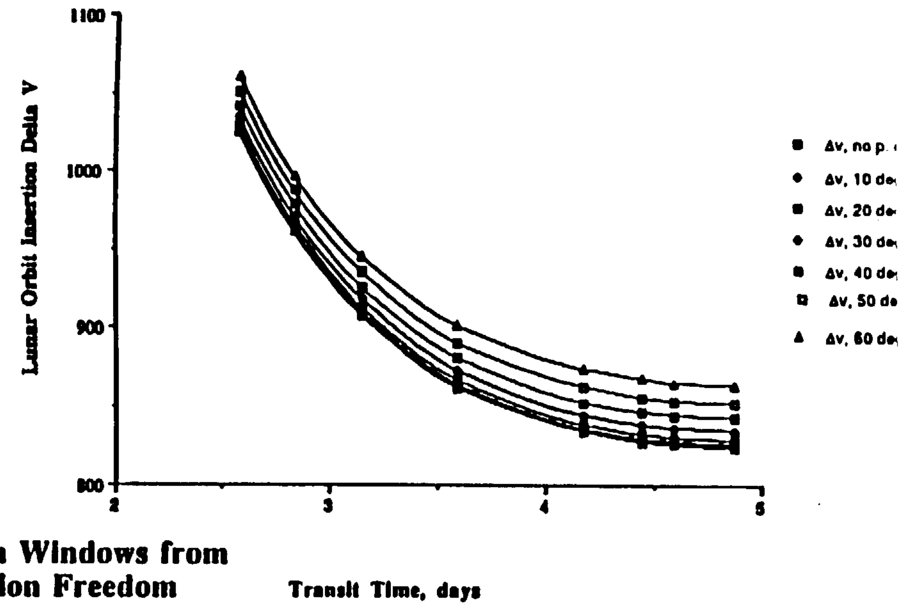
Lunar Analysis

This is a sample of the type of support analysis for lunar trajectory mechanics that has been given to support OEA. It include the timing of translunar trajectories, the lunar orbit insertion ΔV for different arrival asymptotes in degrees per spacecraft, transit time, and the opportunity for lunar transit leaving from the vicinity of the orbiting Space Station.

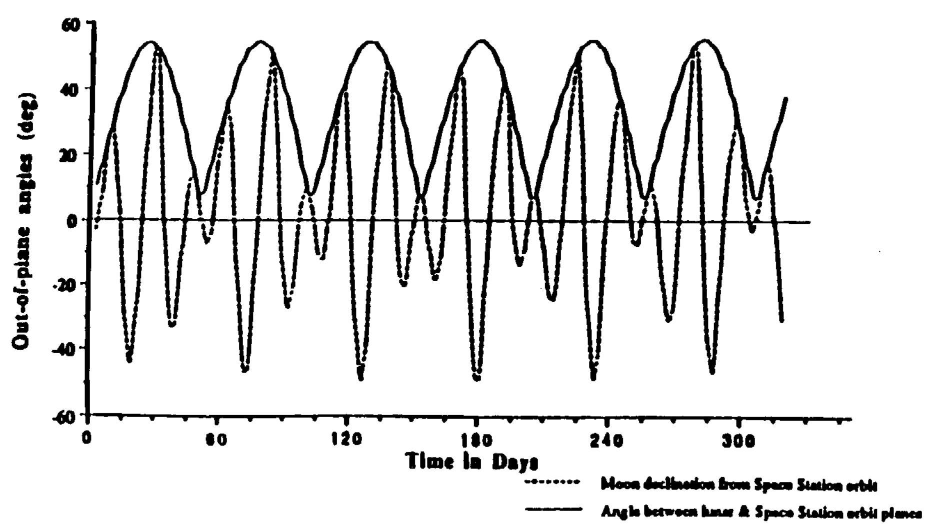
Translunar Trajectory



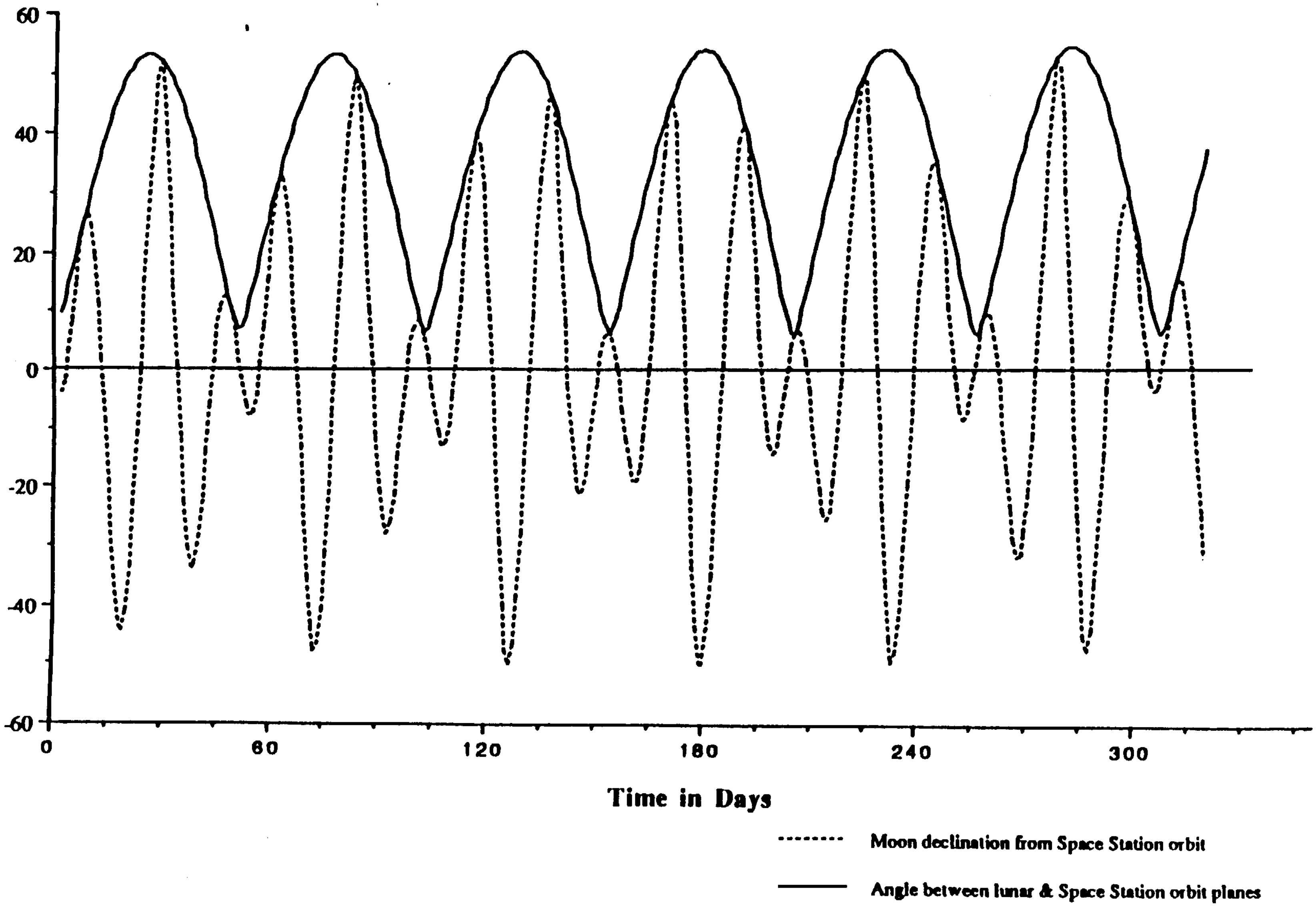
Lunar Orbit Insertion Ideal Delta V (plane change entering lunar orbit)



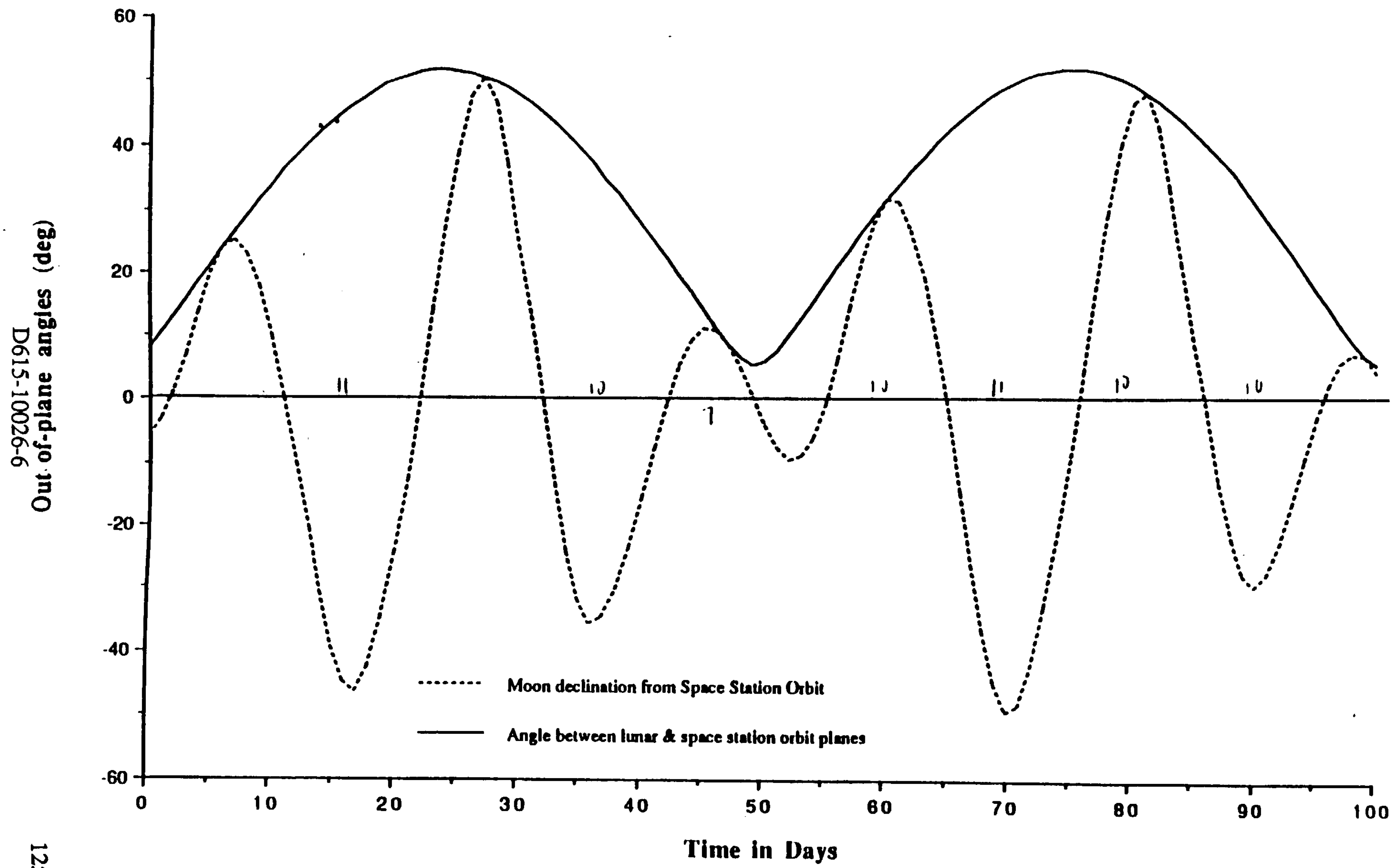
Lunar Launch Windows from Space Station Freedom



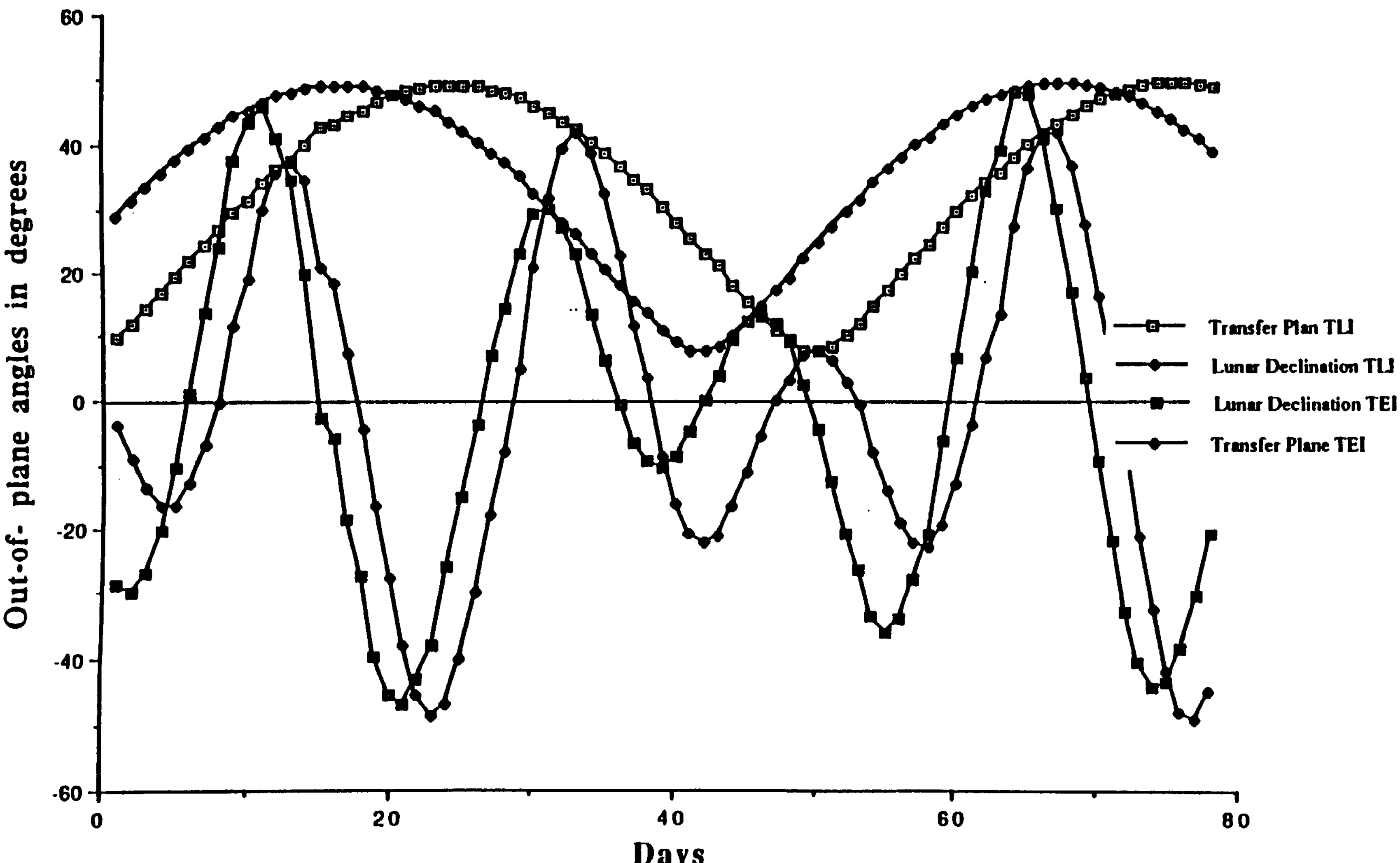
Lunar Launch Windows from Space Station Freedom



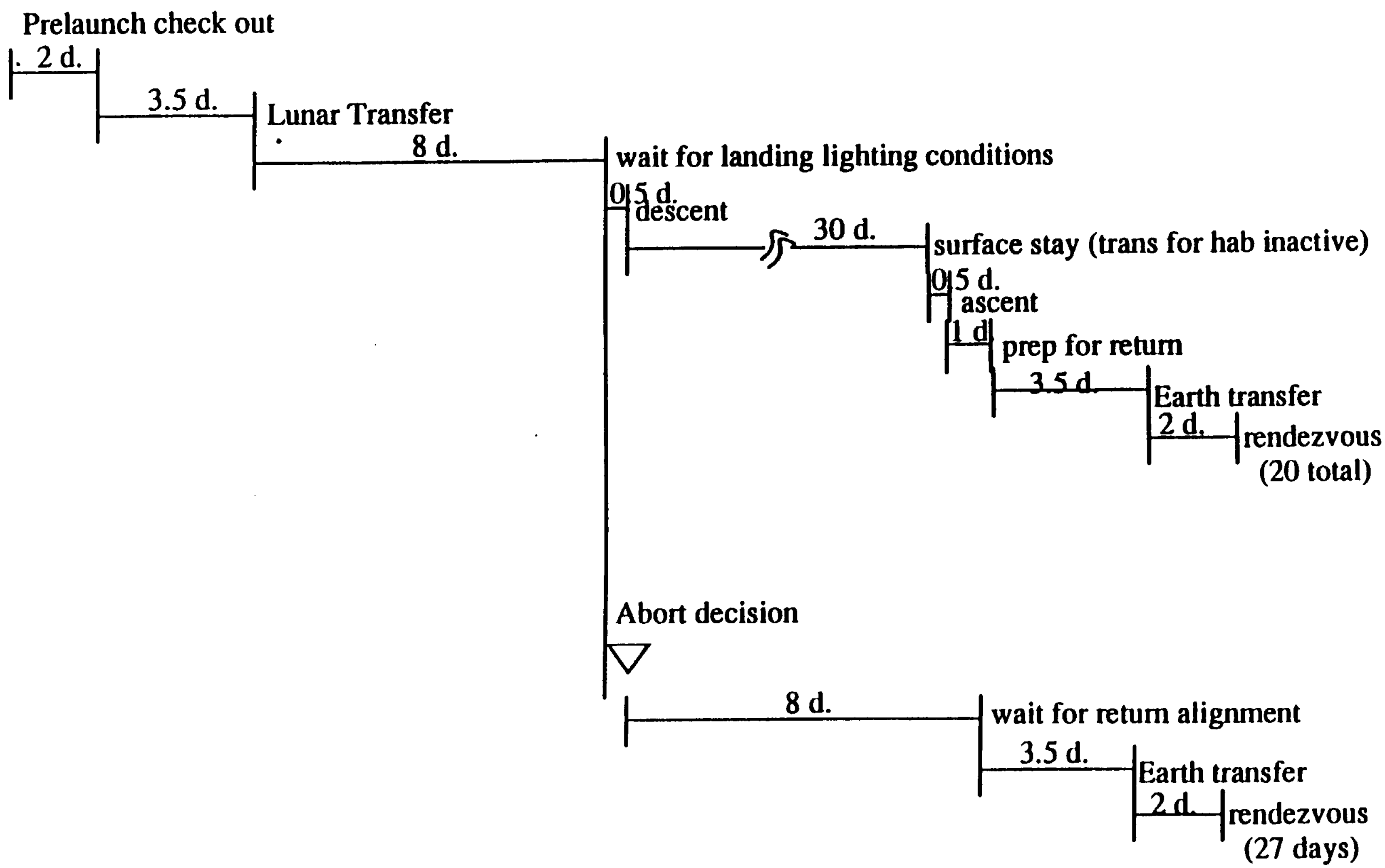
Lunar Launch Windows from Space Station Freedom



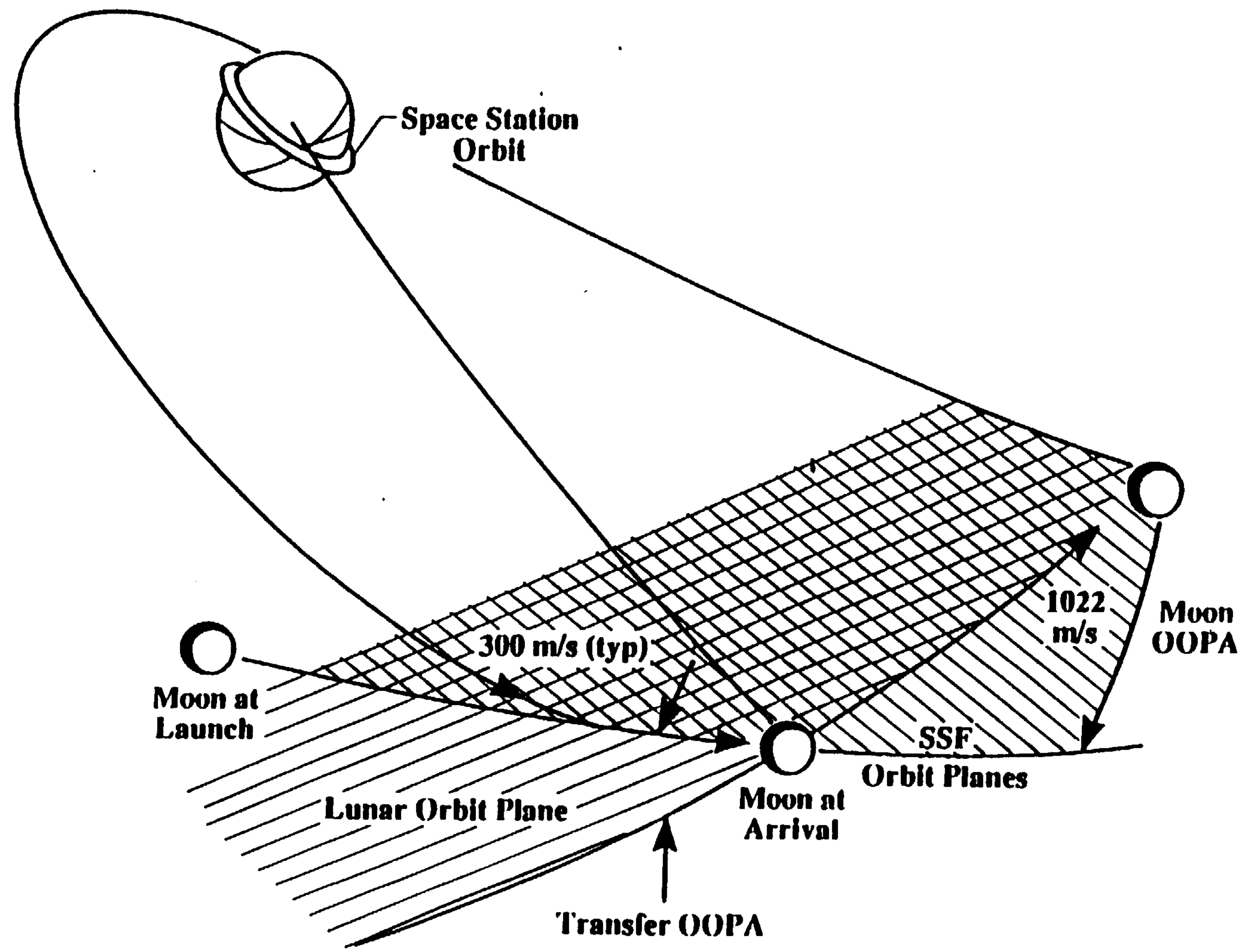
Lunar Launch Windows from Space Station Orbit



Lunar Transfer Hab Timelines
(rough estimate)



Earth-SSF-Moon Transfer Planes



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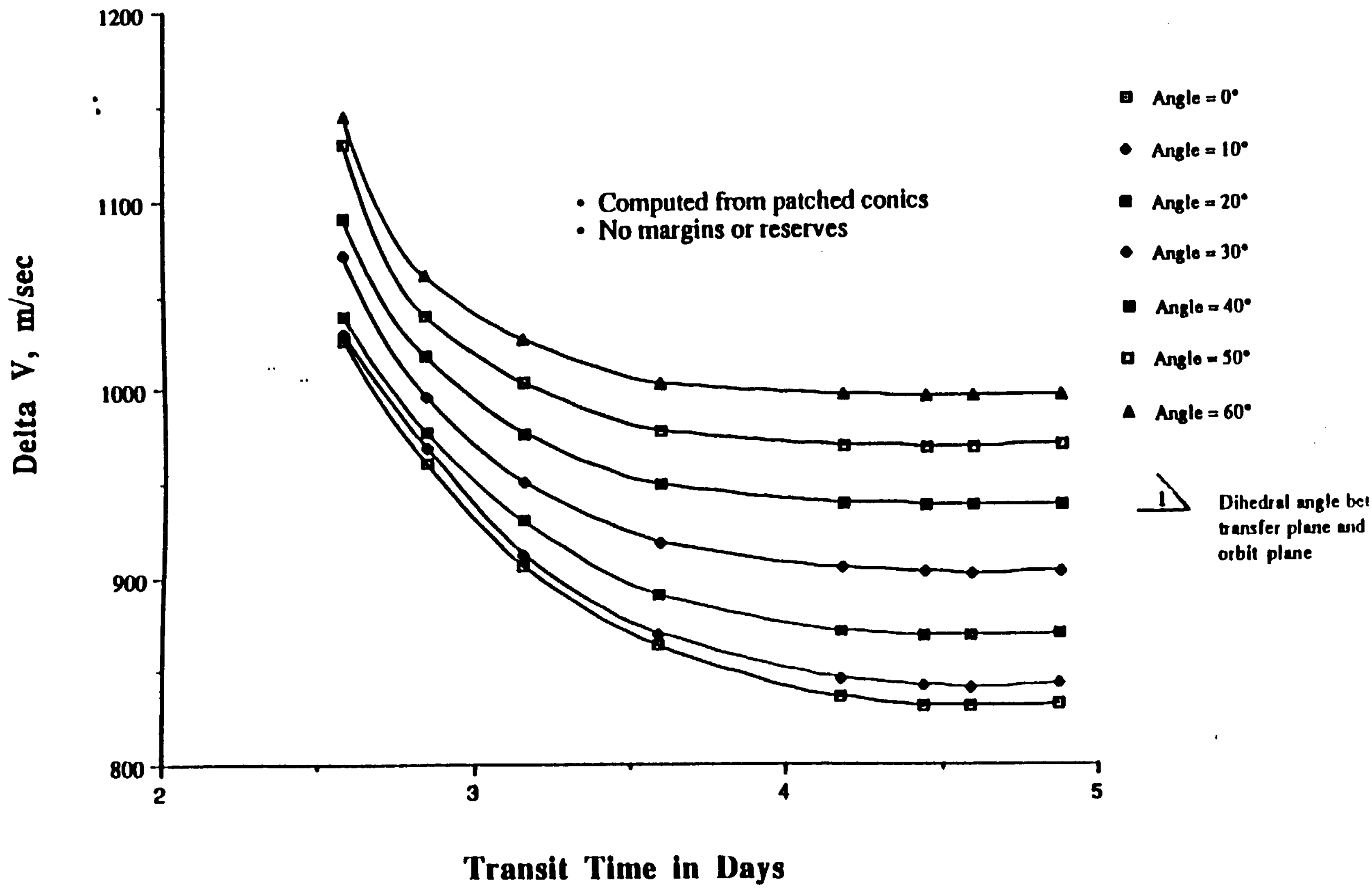
Lunar Orbit Insertion/Departure Delta Vs. SSF to/from Lunar Equatorial Orbit

BOEING

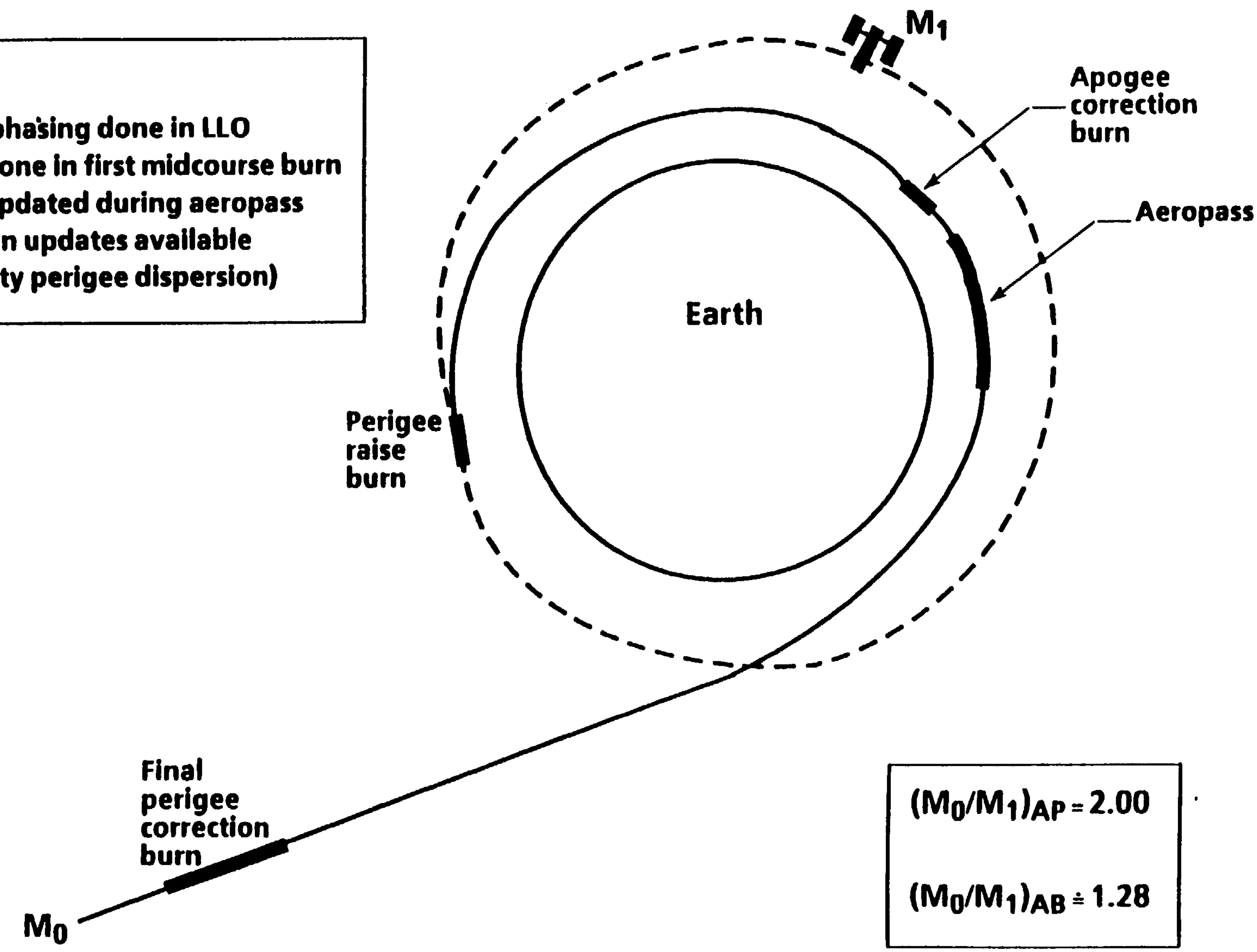


Aerobraking Maneuver Characteristics

BOEING

Ground Rules

- Out of plane phasing done in LLO
- SSF phasing done in first midcourse burn
- SSF phasing updated during aeropass
- GPS navigation updates available
- (± 20M Velocity perigee dispersion)

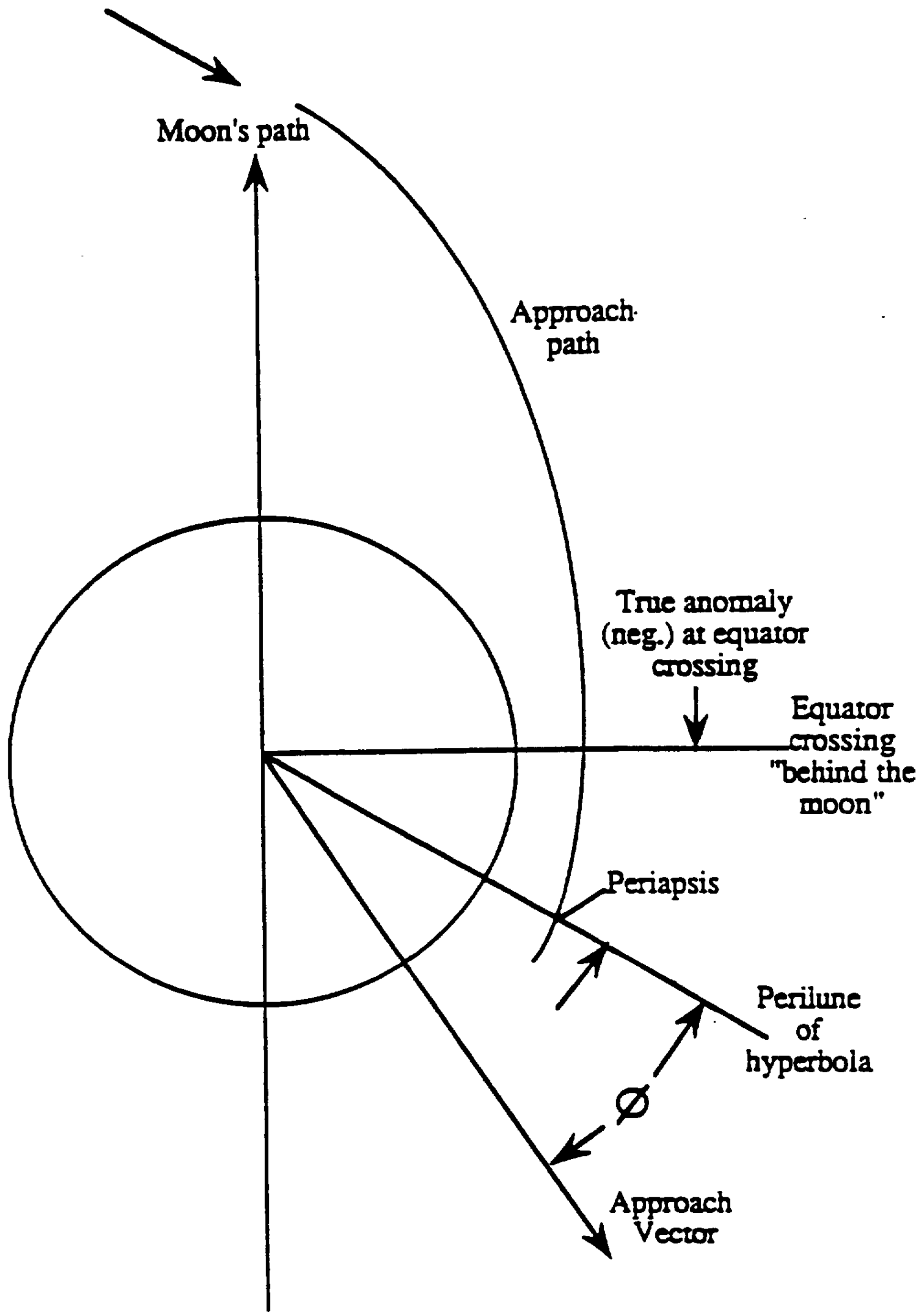


$(M_0/M_1)_{AP} = 2.00$

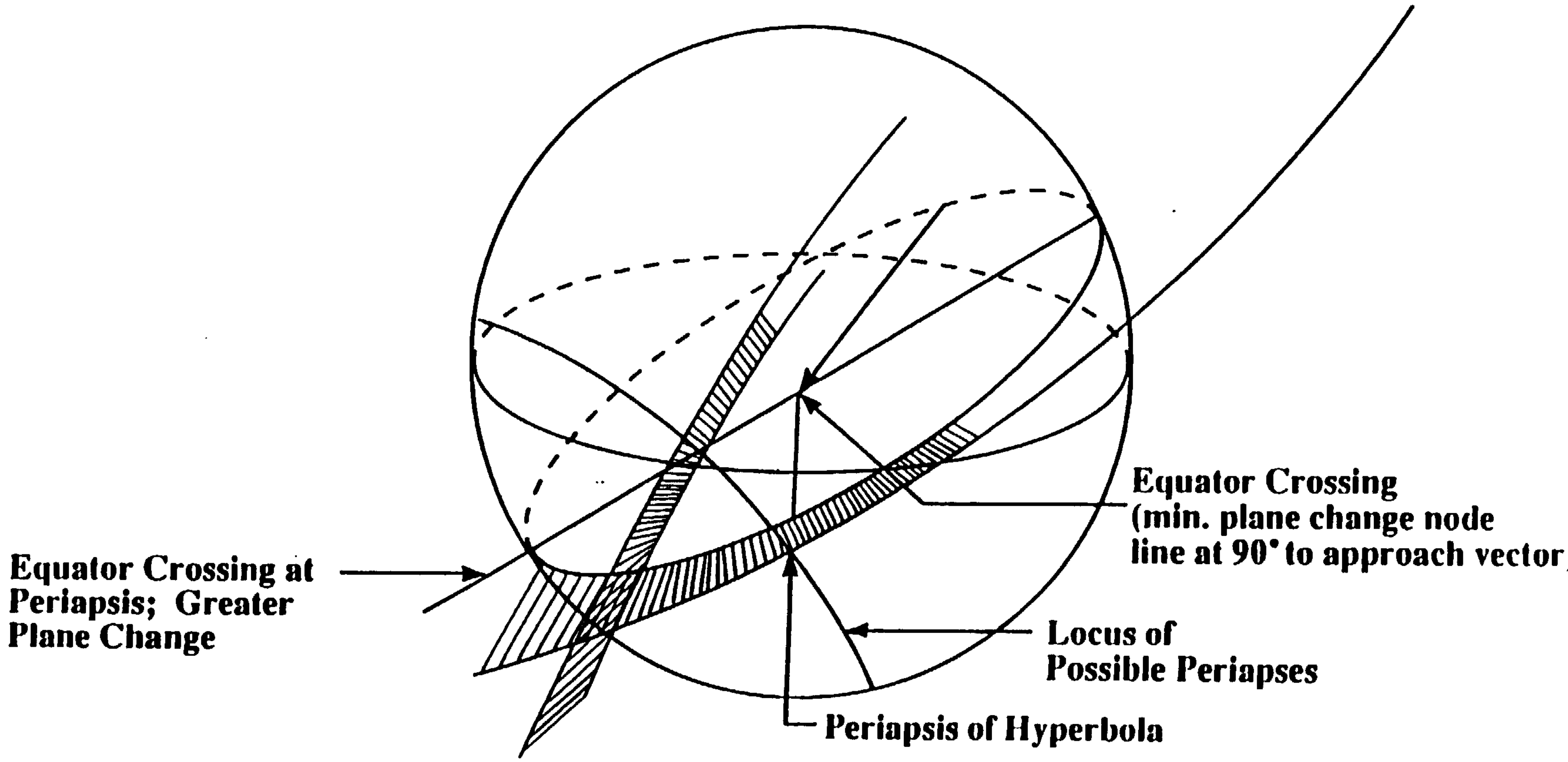
$(M_0/M_1)_{AB} \doteq 1.28$

SSP/2H831/Hab Module/Disk 3/C/256-9

Lunar-Earth Capture Plane Trajectories



Earth Aerocapture Vectors



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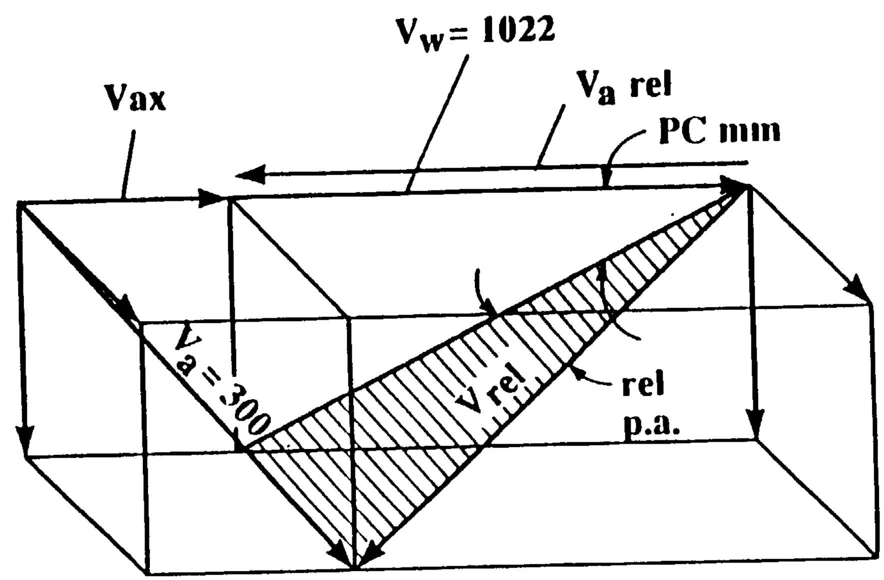
ADVANCED CIVIL SPACE SYSTEMS

Capture Error Correction Vectors

BOEING



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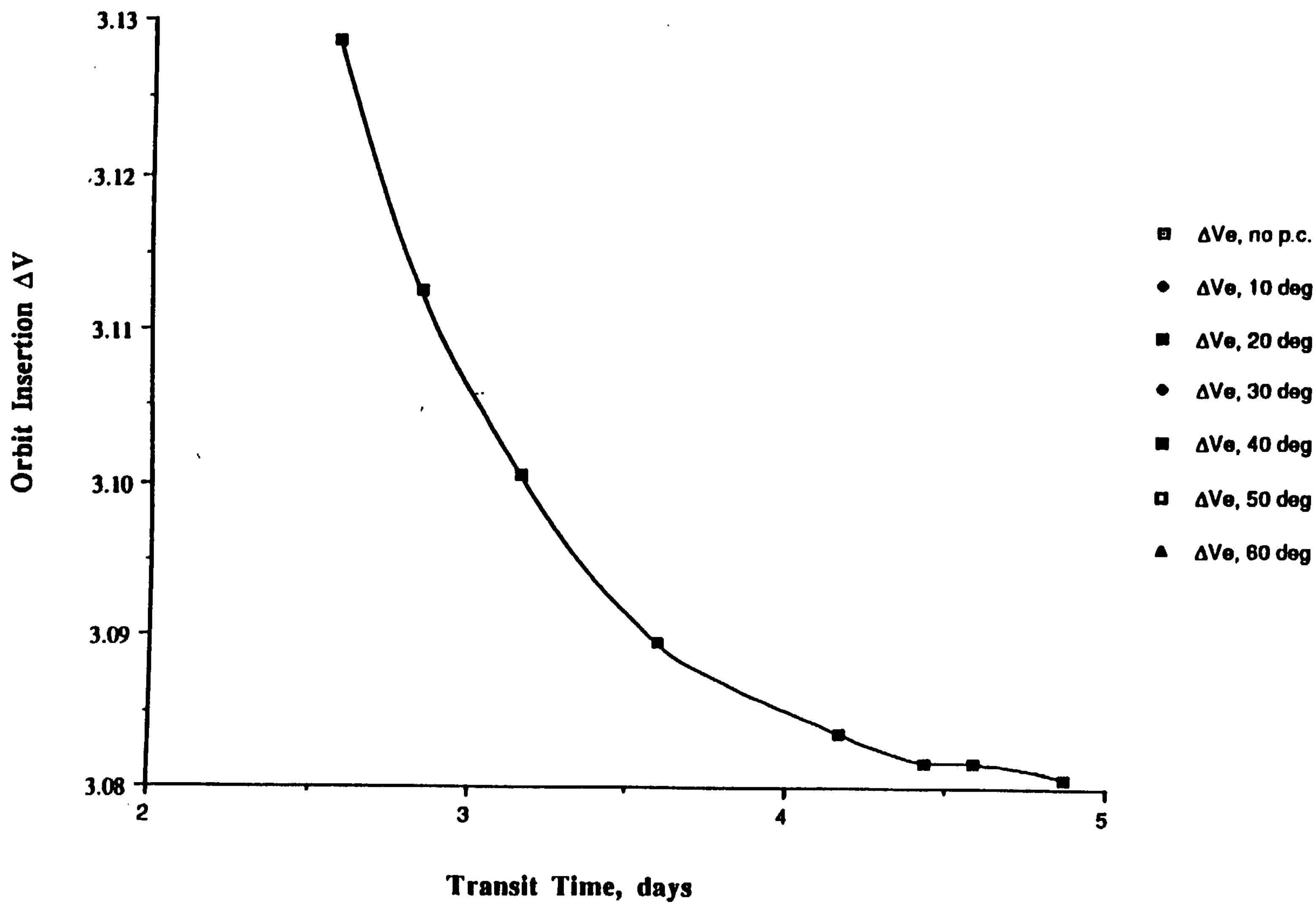
Earth ΔV


3.13
3.12
3.11
3.10
3.09
3.08
Orbit Insertion ΔV
2
3
4
5
Transit Time, days
ΔVe, no p.c.
ΔVe, 10 deg
ΔVe, 20 deg
ΔVe, 30 deg
ΔVe, 40 deg
ΔVe, 50 deg
ΔVe, 60 deg

Recommended Lunar Delta V Budget

Maneuver	Computed ΔV including g loss	2% Reserve	2% for Patched Conic	Recommended Delta V
TLI	3200	X		3265
LOI	860	X	X	895
TEI	990	X	X	1030
Post-Capture	210	X		215

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Lunar Oxygen Delivery Mission Description

	Delta V m/sec	Mass Ratio	Propellant Used	Mass Remaining	
				71,729 kg.	(IMLEO) Resupply = 61,742
TLI	3110	1.95	34,938	36,791	LTV stage 7,902; Aerobrake 2,085; Hydrogen for LEV 23,085; Return propellant 1,203
Lunar pass & L2 arrive	330	1.073	2,516	34,275	
				76,851	(LTV start descent)
Descent from L2	2950	1.884	36,056	40,796	(Landed)
				274,298	(Liftoff)
Ascent to L2	2850	1.844	125,530	148,767	LEV Stage 16,639; Landing legs 1,224; Descent LO2 30,904; LO2 net to L2 95,000; LO2 tank 5,000
L2 depart & lunar pass	330	1.073	765	10,426	
Aerobrake to LEO	200	1.044	438	9,987	LTV stage 7,902; Aerobrake 2,085

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B. Operating Modes and Performance

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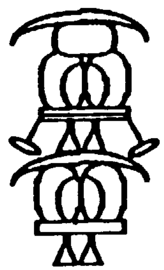
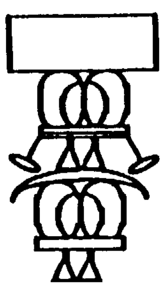
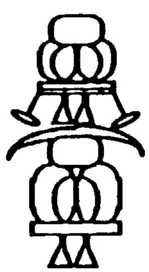
Lunar Modes and Vehicle Options

In examining the architectures, we identified a total of seven lunar vehicle modes shown on the next two pages. The first three are used in all architectures and are shown on the first of the two pages. The next four modes are used in some of the architectures. One is used only in the L2 lunar oxygen architecture, to supply lunar oxygen to the L2 node for use in Mars vehicles. It should be noted that the configuration shown for this particular mode is not a flight configuration but symbolizes the fact that one LTV-sized vehicle can deliver enough hydrogen to L2 to fuel two LTV-sized vehicles to descend to the lunar surface and return to L2 with payloads of lunar oxygen. All of these modes can be implemented with a pair of vehicles: an LTV-like vehicle and an LEV-like vehicle. In examining the modes, we find that both vehicles need to be able to operate with either an LEV-type or LTV-type crew cab. Both need to be able to carry cargo; both need to integrate with an aerobrake; both need to integrate with landing legs; and the larger LTV vehicle needs to integrate with both on the same application. This later case is used only a few times, and can be implemented with expendable landing legs.

Lunar Modes and Vehicle Options

	Mission Mode & Schematic		Application and Rationale	Architectures
1		LTV tandem direct crew, LEO-LS and return	Early crew missions; deferred development of LEV and LEV crew cab.	All
2		LTV tandem direct LEO-LS cargo; booster recovered; lander remains on LS.	Heavy payload capability (45-50 t.); capable of landing campsite intact on single flight	All
3		LTV/LEV crew and cargo LEO-LOR-LS & return; optional lunar oxygen; option to leave LEV on LS in cargo mode.	Most efficient crew mode; efficient with lunar oxygen; lowest lunar oxygen production rate.	All

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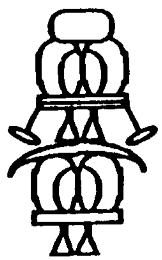
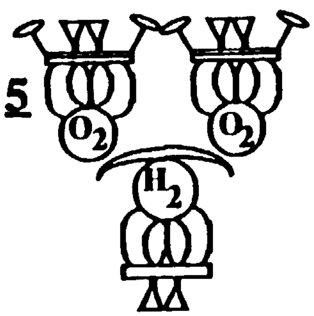
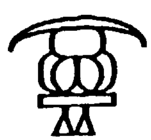
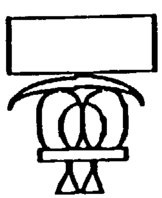
Lunar Modes and Vehicle Options

Mode	Mission Mode & Schematic		Application and Rationale	Architectures
4		LEO-L2-LS crew, rendezvous at L2, with lunar oxygen	As efficient re IMLEO as LOR with lunar oxygen (requires higher lunar oxygen production); simplifies operations for L2 node Mars operations.	L2/Lunar Oxygen; NEP & SEP
5		Lunar oxygen delivery to L2; hydrogen to L2 from LEO	About 1.5 t. lunar oxygen to L2 per t. resupply to LEO; makes for efficient L2 Mars node for cryogenic/aerobraking, or cryogenic all-propulsive conjunction missions.	L2/Lunar Oxygen
6		Crew trips LEO-L2 and return	Mars crews to & from L2 node	L2/Lunar Oxygen; NEP & SEP
7		Cargo trips LEO-L2 with return of LTV	Node and Mars mission cargos to L2	L2/Lunar Oxygen; NEP & SEP

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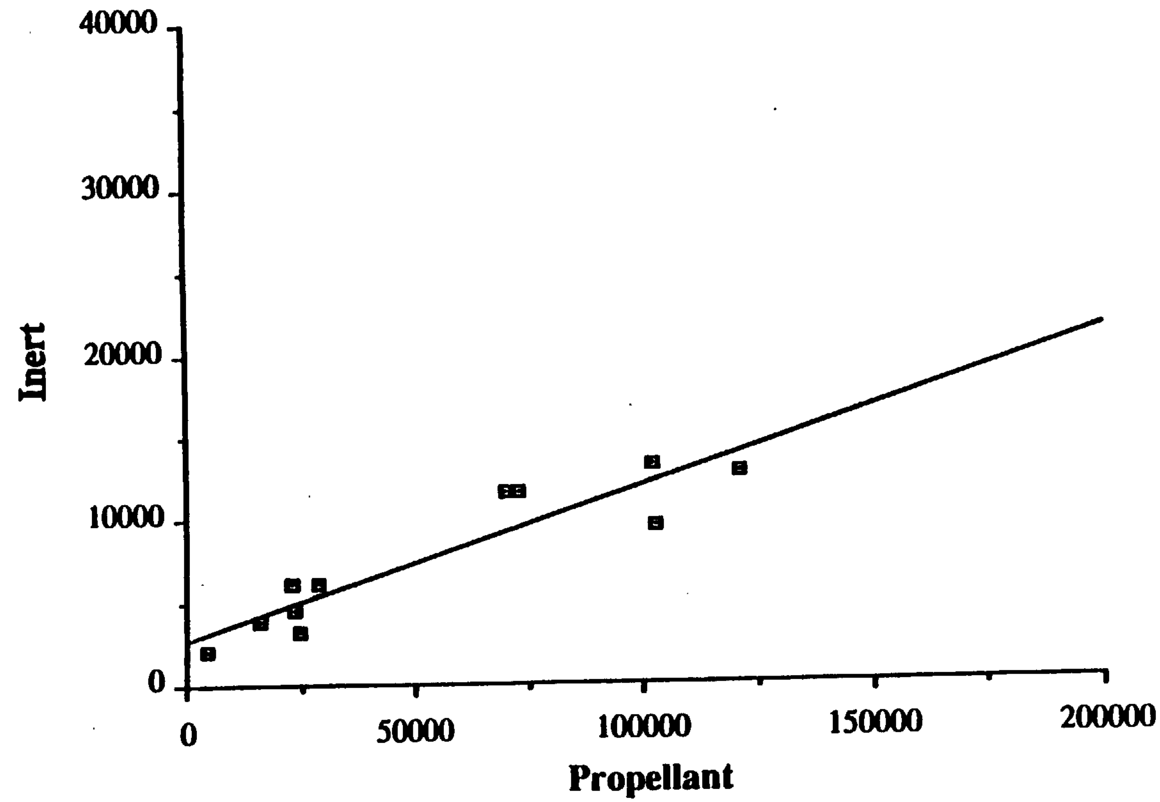
Propellant vs. Inert Mass for Lunar/Mars Vehicles

This parametric curve was developed for all of the STCAEM point conceptual design stages to date. A linear regression of these points results in Mass = 2500 + (0.0875)(Propellant Capacity), in kg. This scaling equation was used to develop lunar mode performance parametrics.

Propellant vs. Inert Mass for Lunar/Mars Vehicles



- Data developed from stages designed in STCAEM study
- Linear regression allows parametric sizing
- Mass = 2500 + (0.0875)(Propellant capacity) [kg]

Lunar Modes Performance

Performance of five lunar modes, for crew rotation and resupply missions, is graphed here. Each mode is characterized for IMLEO, LTV stage propellant loading, and resupply (propellant and cargo). The resupply values do not include mass for propellant or cargo carriers. These vehicles are presumed to be fully reusable; no performance distinction is made between tank replacement and propellant transfer refueling.

The direct tandem LTV mode uses a tandem-staged LTV on a direct mode; there is no lunar orbit rendezvous. The relatively massive LTV crew module (weight assumptions shown on the charts are early estimates accommodating a crew size of 6) is taken to the lunar surface with its radiation storm shelter, as is the Earth return aerobrake. This is not an efficient crew mode; its benefit is that no LEV or LEV crew module is needed, and initial development cost is reduced. If crew trips to the Moon are less frequent than once per year, the development cost savings are indicated as economically more significant than the added launch cost. This is an efficient cargo mode (with the lunar lander LTV left on the lunar surface, until such time as surface refueling with LLOX becomes feasible). A vehicle sized for crew rotation and resupply (based on an integral number of HEI-Shuttle-Z ETO flights for the resupply) can land about 50 t on the Moon.

The direct LLOX mode uses lunar oxygen for the return trip. LOR is the conventional LOR mode, with an LEV crew cab of 3500 kg. The LOR/LLOX mode uses lunar oxygen in the LTV, and it is presumed for the data presented here that the LEV returns to lunar orbit with enough oxygen for its next descent. If lunar trips are infrequent, this may not be practical. In that case, the LOR/LLOX mode can use lunar oxygen only for LEV ascent; its performance is about the same as the direct LLOX mode.

The direct bolo mode uses a rotating tether (bolo) in lunar orbit, as outlined on the following page. The mode is very efficient, but the mass of the lunar orbit bolo is large, and several years' lunar transfer operations may be required to emplace it. From a mass and operations standpoint, then, the bolo mode appears to provide a costly and only marginal performance benefit over the use of LLOX.

Lunar Modes Performance

Crew Mission, 1 t. Payload Returned

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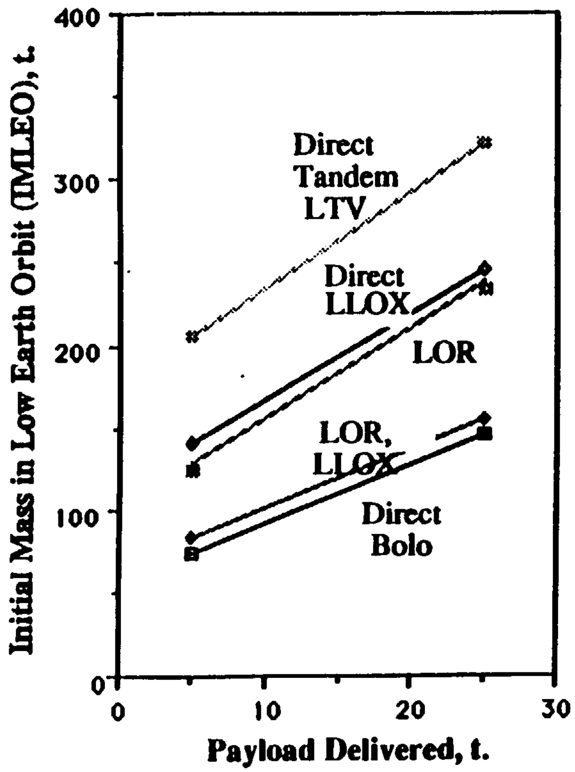
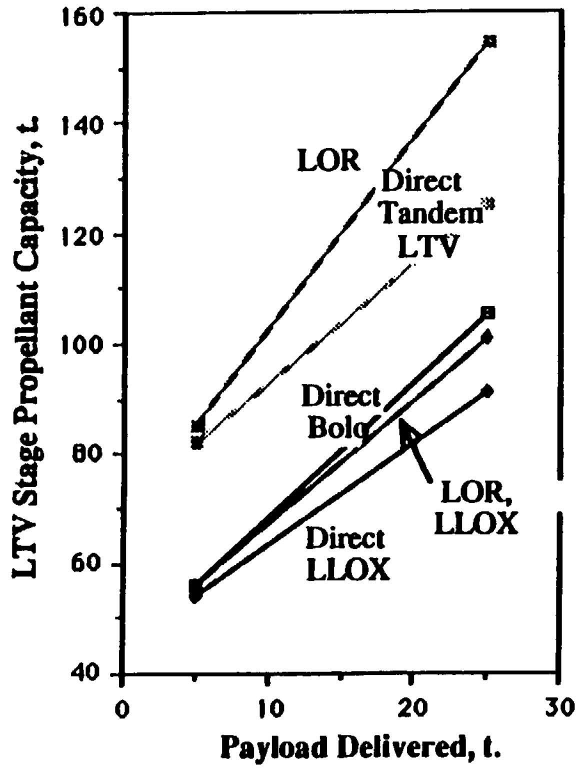
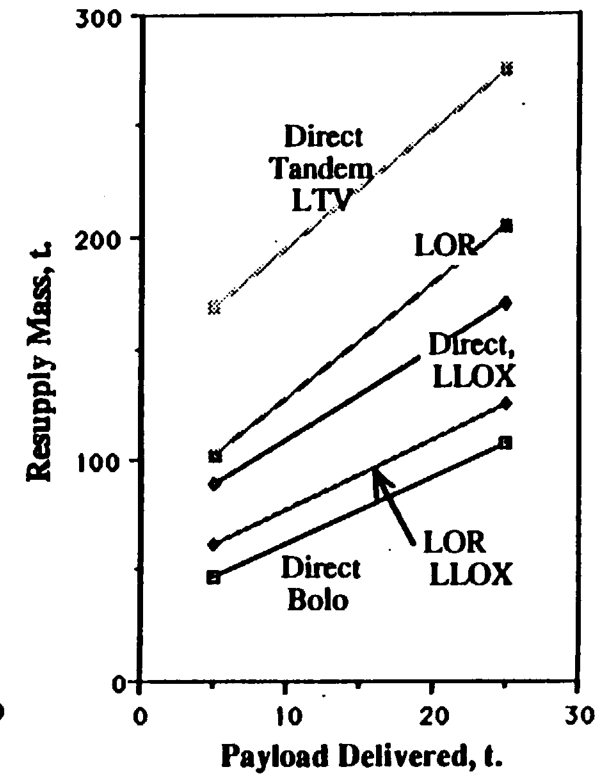
IMLEO Performance



Propellant Loading



Resupply Performance



Note : LTV crew cab mass 8.05 t; LEV crew cab mass 3.5 t.
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Lunar Modes Performance

Cargo Delivery Mission

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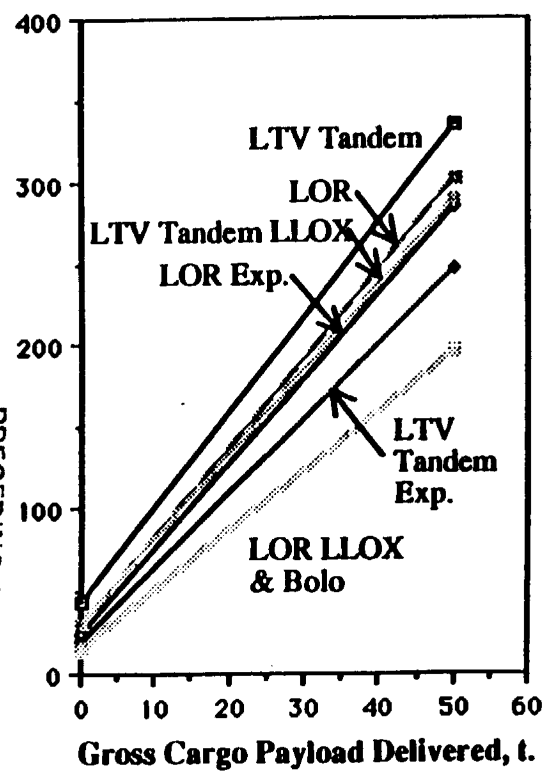
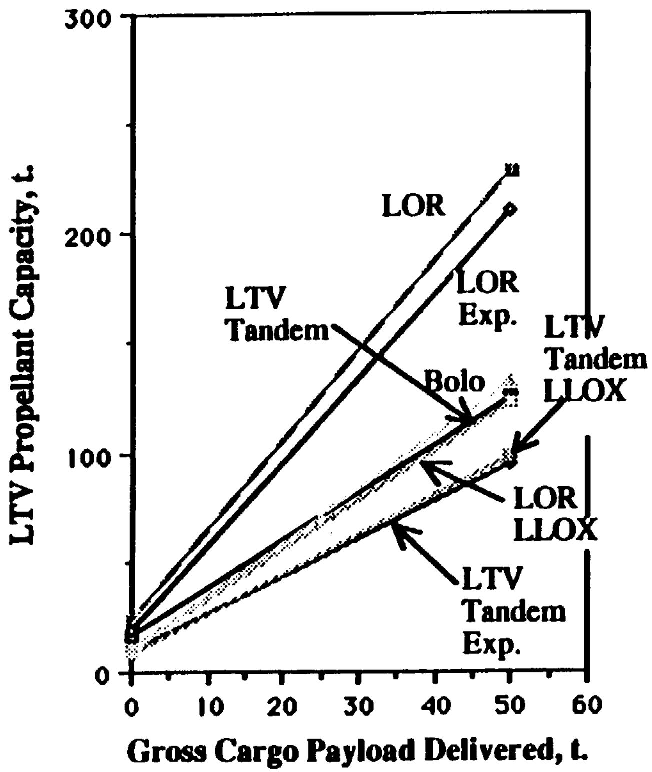
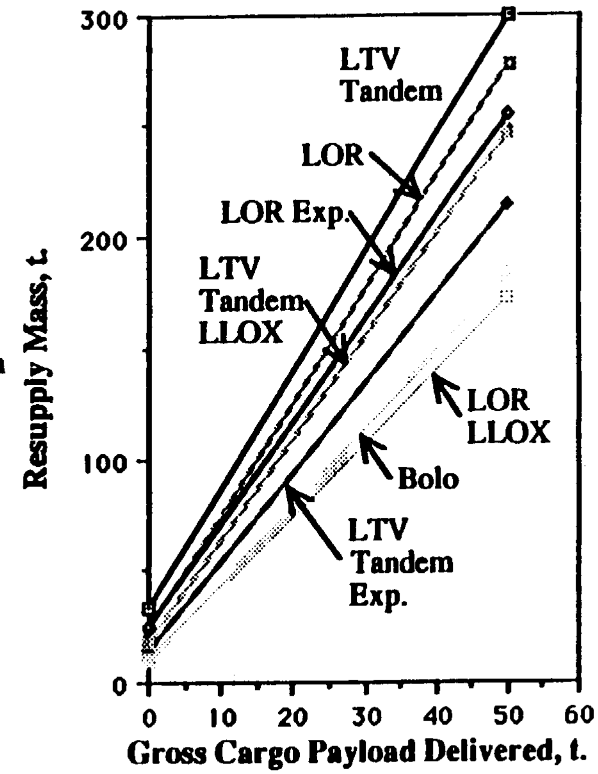
IMLEO Performance



LTV Capacity



Resupply Performance



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Lunar Transportation Family
Systems Required for Varying Missions

11 different mission modes have been identified and are matrixed with the 5 major lunar transportation family systems. This chart identifies the number of each system required to accommodate the mission modes identified. The last column shows the propellant required from earth for each mission mode, which points out the efficiencies of the LLOX cases once LLOX is available. In the NEP/SEP Crew Delivery mode, the propellant capacity required is not substantially larger than the 25 t LEV propellant capacity. Therefore, this may be a rationale to begin thinking about larger LEV stages that can fulfill NEP/SEP crew delivery as well as extend "hopper" distances on the lunar surface.

Lunar Transportation Family

Systems Required for Varying Missions

Mission Mode \ System	110 t Propulsion Stage	25 t Propulsion Stage	Transfer Hab	Crew Cab	Aerobrake	Propellant From Earth (kg)
Tandem Direct Cargo	2	---	---	---	(1)	201,553
Tandem Direct Crew	2	---	1	---	(1)	163,093
Tandem Direct Large Cargo	2	---	---	---	(1)	201,553
LOR Cargo	1	1	---	---	1	130,865
LOR Crew and Cargo	1	1	1	1	1	123,181
LOR Cargo using LLOX	1	1	---	---	1	65,109
LOR Crew using LLOX	1	1	1	1	1	73,274
Cargo Delivery Through L_2	2	---	---	---	1	159,153
Crew Delivery Through L_2	2	---	1	1	1	159,145
NTR Resupply	1	---	---	---	1	42,021
NEP/SEP Crew Delivery	1	---	1	---	1	33,556

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IV. Element Descriptions

A. LTV/LEV Components

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IV. Element Descriptions

Element descriptions for the lunar transportation family included in this document are a listing of the LTV/LEV components, trade studies and mass analyses of the transfer and excursion modules, ACRV (MCRV) modifications required to fulfill lunar operations, the aerobrake shape and L/D to be used, and some costing methods and results.

A. **Component listings**, assumptions and sizing criteria are included for the LTV, LEV, ACRV and the service module (Apollo command module derivative). This information is provided to give an overview of the major components of the lunar transportation family and their related subsystems.

B. An LTV/LEV habitat trade **module study** was conducted to size crew modules for varying crew sizes and mission durations. Two types of transfer modules were evaluated, an aerobraked module and a direct entry (Apollo-type) module, as well as a single module concept for transfer and excursion (direct entry at Earth) and excursion modules. Crew sizes of 2, 4, 6, and 8 for transfers of 24 days and surface stays of 1, 14, 28 and 42 days. Sizes for these 36 modules were generated from historical spacecraft data, and mass statements were generated from SSF and STCAEM estimates. Results of this trade study provide good estimates as to the size and mass of lunar crew modules.

A trade study was also performed to determine what point it becomes more mass efficient to have a separate surface hab along with an excursion module, if a base is not available and missions of the excursion/exploration class are being performed. Results show that 3-8 days is the crossover point.

C. The SSF **ACRV** was originally believed to be easily adaptable to small scale lunar mission, which would allow the use of "existing" hardware. For the small scale program, the ACRV was to be used for the reentry phase back at Earth. However, the ACRV currently envisioned for SSF will not fulfill lunar mission needs because of its size. The internal volume is extremely limited since it is designed for a 6 hour mission rather than 7 to 24 day missions. In order to provide sufficient volume for crew operations and equipment storage, the interior volume would need to be increased approximately 250% (giving the same amount of free volume as the Apollo Command Module). Increasing the volume requires major structural modifications and the resulting craft would be unlike the current ACRV. Therefore, a more appropriate use for the ACRV is as a crew module for the LTV in a direct-to-surface (tandem direct) mission scenario. Prior to Earth reentry, the crew transfers to the ACRV and separates from the LTV, which is expended.

D. The **aerobrake** to be used in the lunar transportation family is envisioned as being an early version of the Mars low L/D (L/D = 0.5) shape. The idea is to take the symmetrical center portion of the hyperboloid shape and fly at an L/D of about 0.25. This symmetrical portion of the Mars shape is about the right size to accommodate the LTF without having to drop tanks. Another variation to this concept is to have standard "additions" that can be added to the outer rim of the brake to accommodate growing mission needs. When Mars comes into the picture, an "addition" can be fabricated to accommodate these missions.

Heating analysis was performed on this shape, and it was found that the ballistic coefficient is very low in comparison with Mars and the heating temperatures only reach about 1870°K at the stagnation point.

E. **Costing** for the lunar transportation family is being calculated for both hardware cost, using the Boeing Parametric Cost Model (PCM), and life-cycle cost, using a model

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developed by Madison Research. Preliminary costing date from PCM is included in this document, and the life-cycle cost data is in progress and will be reported in later documents (IP&ED updates, final report, technical directives, etc.).

Components for Direct Flight, Tandem LTVs

LUNAR TRANSFER VEHICLES:

* Booster
  - Fuel
  - Aerobrake
  - Electronic Autonomous Control
  - ACS
  - Structures
* Lander
  - Fuel (ascent and descent)
  - Electrical connection to ACRV
  - Surface Access (up and down)
  - Structures
  - ACS
  - Fitted with Landing Legs
  - Must contain both the ACRV and Service Module

INTERSTAGES:

- Structures
- Disconnect Mechanisms

LTV and LEV Sizing

Assumptions:
- Medium Scale Scenario
  - Lunar Oxygen Production
  - Permanent Science Base
  - Crew of 6
  - Overnight Stays until Base is completed
  - Yearly Flights
- Use LLOX to refuel LEV on Lunar Surface

LEV:
- Capable of delivering 15 metric tons of payload to the Lunar Surface and returning 1 ton in the crew-mode.
- Ascent and Descent propellant loads were calculated from split payload and fueling equations.
- LLOX was used for the oxidizer in both phases; the necessary hydrogen must be delivered from the Earth.
- The LEV tanks are filled in LEO, rather than in LLO by the LTV.
- The propellant tanks are capable of containing 35 metric tons.

LTV:
- The tanks must be capable of holding all the required propellant for the worst-case scenario (in terms of the propellant loading).
  - The LEV must be brought from the Earth.
  - The LEV delivers 15 tons of payload to the surface and returns 1 ton to lunar orbit.
  - The LTV returns to the Earth.
  - Lunar Oxygen is used for the ascent and descent phases.
- LTV tanks can hold 110 metric tons of propellant.

Components for Direct Flight, Tandem LTVs

SERVICE MODULE:
- Repressurization air for surface operations
- Excess water
- Excess waste
- Extended breathing air
- Fuel (may house the ascent fuel)
- ACS
- Thermal Control System
- Shielding
- Power System (fuel cells, deployable solar arrays)
- High gain antenna
- Communications (some for Earth and EVA)
- Science Package
- Payload Storage (Mass is separate) ???
- Most of the mass has been included in the resizing of the ACRV

ACRV: (See <u>Resize</u> spreadsheet)
- Crew
- Thermal Control System
- ECLSS
- Hygiene
- Medical Equipment
- Experiments
- Payload
- EVA Equipment (cleanup and storage)
- Control System
- Visibility and Orientation

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Service Module for the ACRV

ASSUMPTIONS:

APOLLO SERVICE MODULE	
Mass (dry)	5230 kg
Mass (with propellant)	25000 kg
Height	7.44 m
Diameter	3.95 m
Propellants	
SPS Fuel	7170 kg
SPS Oxidizer	11460 kg
RCS	620 kg

Major Subsystems
- Electrical Power
- Environmental Control
- Reaction Control
- Service Propulsion
- Telecommunications

Reference: Apollo Spacecraft Reference (Lunar Module), Grumman Aerospace Corp., 19??.

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B. Habitation Modules

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LTV/LEV Habitat Module Study

The LTV/LEV habitat module study was conducted to determine the most reasonable crew modules that can be used in the LTV/LEV system. The study encompassed transfer, both aerobraked and direct entry, and excursion modules as well as a combination transfer/excursion direct entry module.

LTV/LEV Habitat Module Study

Goals

- Determine configuration envelopes for LTV/LEV habitat modules
- Develop mass statements for each configuration

Groundrules

- Crew sizes of 2,4,6 and 8
- Surface stays of 1, 14, 28 and 42 days
- Round trip time of 7 days - 24 day free return abort
- Crew volumes extrapolated from historical data
- Mission modes
  - Transfer module in conjunction with excursion module
    - aerobrake at Earth
    - direct entry at Earth
  - Transfer/excursion module
    - single module - direct entry at Earth
  - Excursion module in conjunction with transfer module

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LTV/LEV Hab Modules

Assumptions

- Sized for 24 day free-return abort in the worst-case
- Stored O2 for the breathing with regenerative molecular-sieve beds for CO2 removal
- No "hygiene" water allocated for showers, washers, galleys, etc.
  Stored H2O at 2.0 kg/man day used for drinking, food preparation, and sponge bath
- Food is all shelf-stable - 1.25 kg/man day
- 10 kg/cm2 radiation shielding for shelter in addition to approximately 2-5 g/cm2 of protection provided by skin structure and onboard equipment - 48 hour nominal duration
- No refrigerators, freezers, personal hygiene compartment allocated
- Minimal exercise equipment - "bungee cord" type
- ACS provides cabin air leakage make-up and 3 cabin repress. recharges
- Human waste and urine storage - no urine processing
- Power supply - solar arrays with batteries during lunar night, backup and aeromaneuver periods
- 15% mass growth

Operating Modes

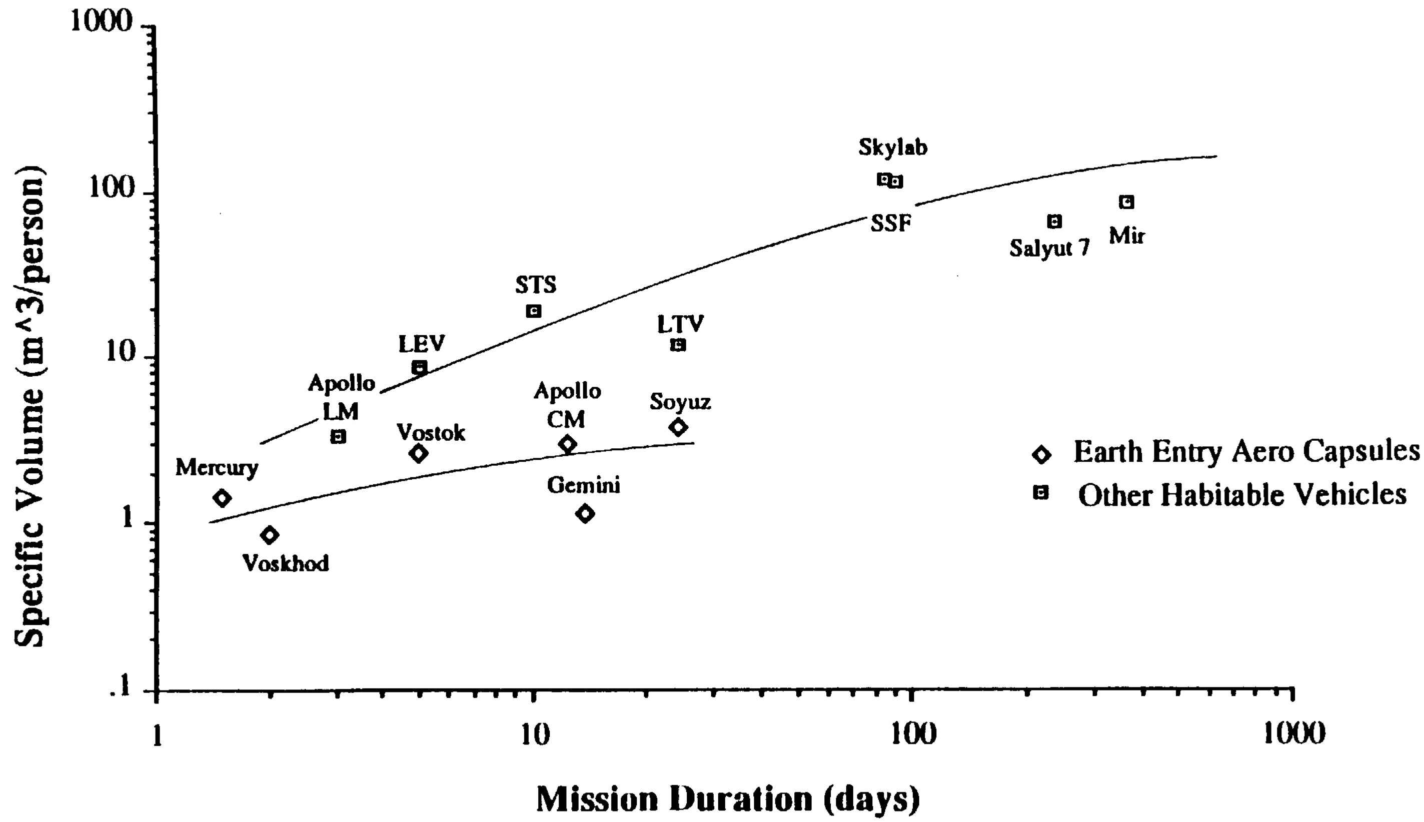
- Transfer module in conjuction with an excursion module
- Transfer/excursion module (1 module) both aerocapture and direct entry

LTV/LEV Crew Volume Guidelines

This graph, which shows historical spacecraft total pressurized volume, was used as a guide for determining optimum crew volumes required for various mission durations. The LTV and LEV modules are plotted on this graph.

LTV/LEV Crew Volume Guidelines

Historical Spacecraft Total Pressurized Volume Data



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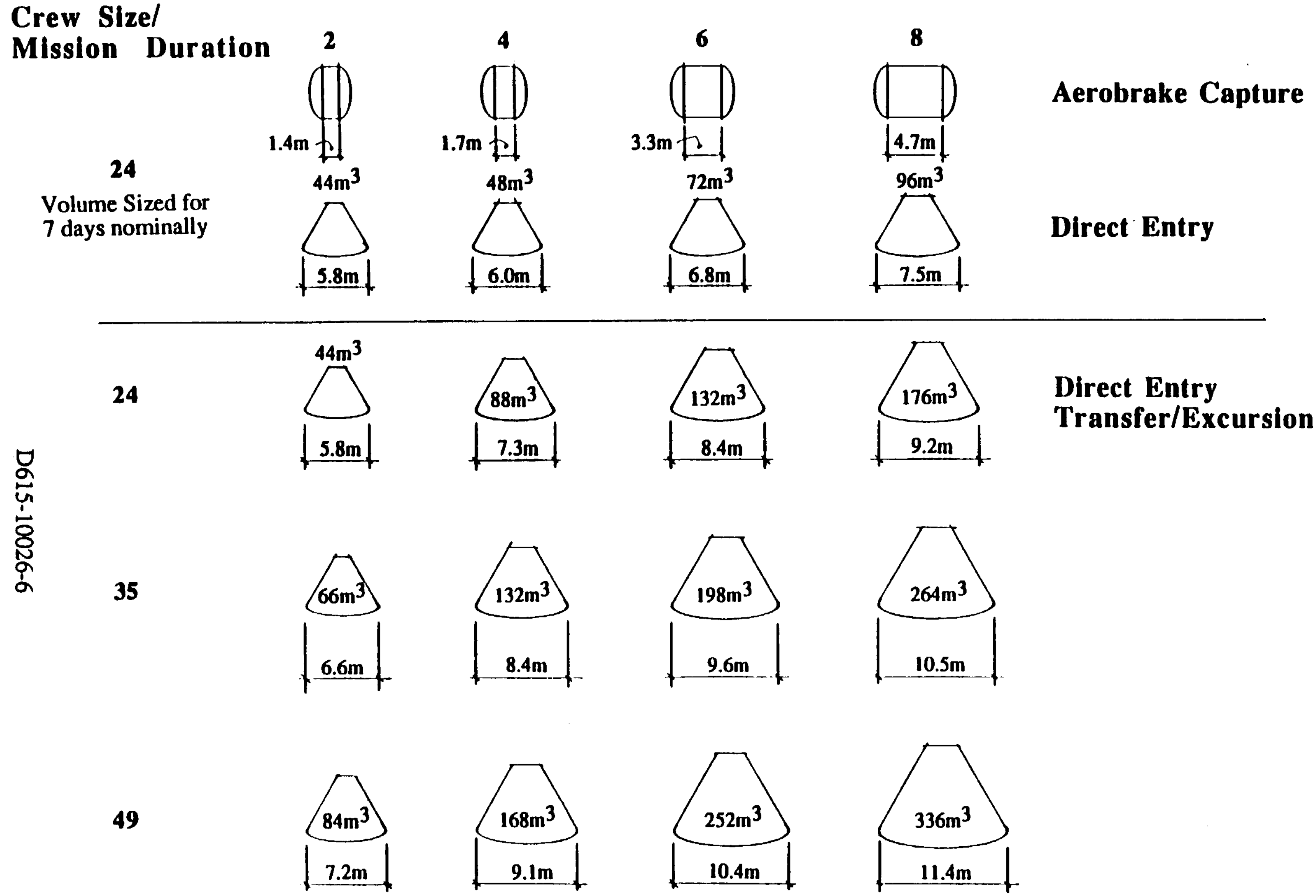
Lunar Transfer Modules Configuration Envelopes

This chart shows the relative sizes of the 3 transfer modes studied: transfer aerobraked, transfer direct entry, and transfer/excursion direct entry. The direct entry shapes are all "Apollo type".

Lunar Transfer Modules
Configuration Envelopes



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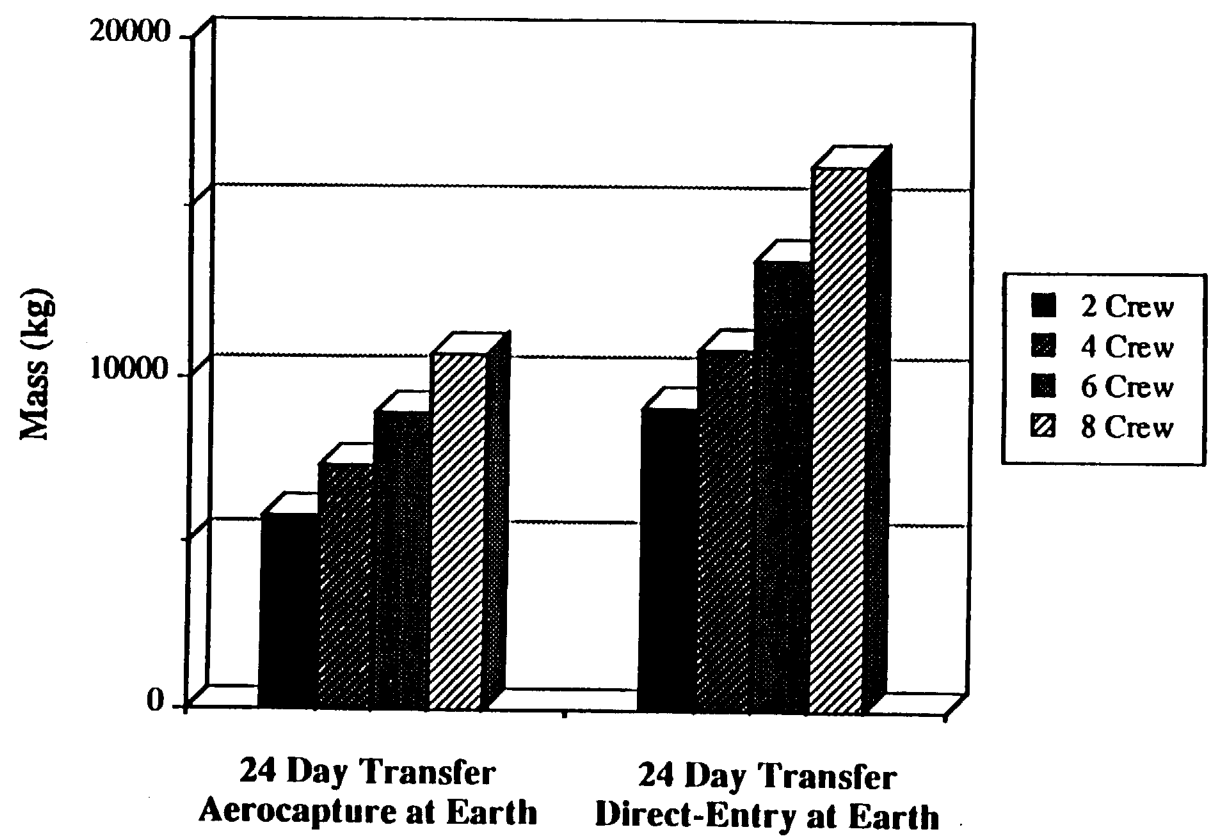
Lunar Transfer Module Mass Summary

Shown on this chart are the relative masses of the 2 transfer modes, aerocapture and direct entry. The direct entry is naturally more massive because of the inefficient pressurized shape and the extra equipment required for landing.

Lunar Transfer Module Mass Summary



- **Operates in conjunction with an excursion module**
- **Sized for a 24 day free return abort**
- **"Direct-entry" module is an "Apollo" shape**

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Lunar Transfer Crew Module

24 day duration

Crew Size	2	4	6	8
Habitable Volume (m3)	44	48	72	96
ECLSS Total	1,272	1,317	1,361	1,406
Atmosphere Revitalization System (ARS)	242	242	242	242
ACS (tanks & 1/2 nec. SSF equip.)	274	290	306	322
Atmos. Composition Monitor Assembly	125	125	125	125
Thermal Control/Temp. & Humidity Control (1/2 SSF - av. air equip.)	479	479	479	479
Potable Water and Storage System	29	58	86	115
Fire Detection and Suppression System	123	123	123	123
Structure Total	1,472	1,575	1,978	2,442
End cones (2)	266	266	266	266
Berthing ring/mechanism (1)	139	139	139	139
Berthing interface plate (1)	90	90	90	90
Cylinder primary structure	295	322	467	608
Cylinder secondary structure	352	388	579	765
Standoff/utililities/ distribution	96	96	123	220
Hatches (2)	134	134	134	134
Windows (4)	60	60	60	60
Couches/sleepers	40	80	120	160
Command/Control/Power Total	524	762	802	842
ECWS (1/2 SSF)	22	220	220	220
DMS/audio - visual	280	280	280	280
Fault detection and isolation	40	40	40	40
Power system (solar arrays, batteries, onboard equipment)	132	172	212	252
Lights (1/2 SSF)	50	50	50	50
Man-Systems Total	417	457	497	537
WMS/waste storage	95	95	95	95
EVA suits/space closure balls	272	312	352	392
Medical equipment (1/2 surf. equip. mass)	50	50	50	50
Consumables Total	498	852	1,278	1,704
Food and packaging	120	240	360	480
Atmospheric make-up and 3 represses (20% reserve)	206	269	403	538
Other (clothes, hygiene equip., etc.)	28	55	83	110
Potable water	144	288	432	576
Other Total	1,083	1,691	2,299	2,973
Personnel and effects	180	360	540	720
Equipment spares	100	104	108	112
Tools	25	25	25	25
Radiation shelter	778	1,202	1,626	2,050
Mass Growth Total (15%)	553	617	696	784
Total Module Mass	5,819	7,271	8,911	10,688

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

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Lunar Transfer Crew Module
Direct Entry

24 day duration

Crew Size	2	4	6	8
Habitable Volume (m3)	44	48	72	96
ECLSS	1,272	1,317	1,361	1,406
Structure	1,766	1,890	2,374	2,930
Command/Control/Power	524	762	802	842
Man-Systems	417	457	497	537
Consumables	498	852	1,278	1,704
Personnel and Effects, Spares, etc.	305	489	673	923
Radiation Shelter	778	1,202	1,626	2,050
Earth Entry Heat Shield	2,008	2,149	2,760	3,358
Earth Recovery Equipment	454	547	682	825
Mass Growth (15%)	1,020	1,180	1,441	1,712
Total Module Mass	**9,042**	**10,845**	**13,494**	**16,287**

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

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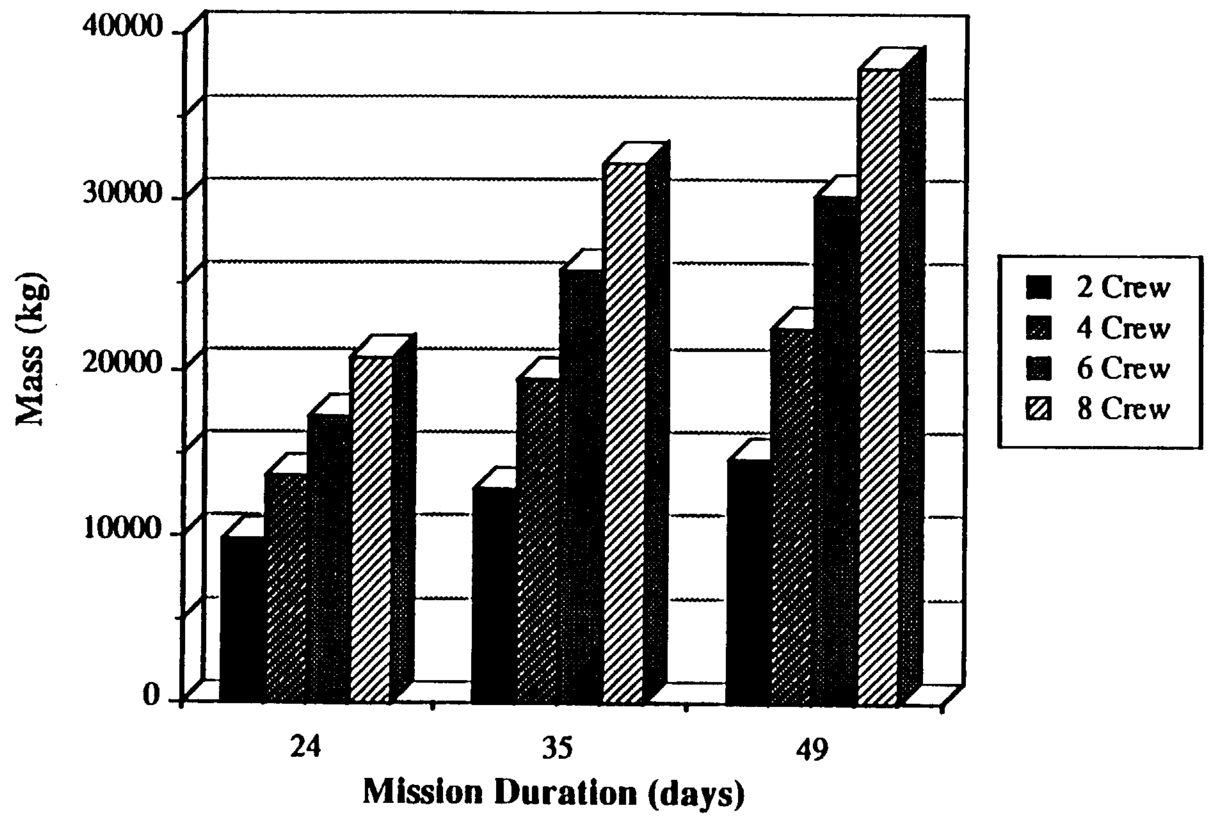
Lunar Transfer/Excursion Direct Entry Module Mass Summary

Shown on this chart are the relative masses of the direct entry transfer/excursion modules for surface stays of 1, 14, 28 and 42 days with a 7 day transfer time.

Lunar Transfer/Excursion Direct Entry Module Mass Summary



- Single module ("Apollo" shape) serves as transfer and excursion module
- 24 day duration includes 1 & 14 day surface stays and is sized for a 24 day free return abort worst case
- 35 day duration includes a 28 day surface stay
- 49 day duration includes a 42 day surface stay

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Lunar Transfer/Excursion Crew Module
Direct Entry

24 day duration*

1 day and 14 day surface stays

Crew Size	2	4	6	8
Habitable Volume (m3)**	44	88	132	176
ECLSS	1,272	1,317	1,361	1,406
Structure	2,321	3,145	3,901	4,838
Command/Control/Power	524	762	802	842
Man-Systems	417	457	497	537
Consumables	498	852	1,278	1,704
Personnel and Effects, Spares, etc.	305	489	673	923
Radiation Shelter	778	1,202	1,626	2,050
Earth Entry Heat Shield	2,008	3,181	4,212	5,052
Earth Recovery Equipment	487	684	861	1,041
Mass Growth (15%)	1,109	1,544	1,914	2,284
Total Module Mass	**9,719**	**13,633**	**17,125**	**20,677**

* 1 day surface stay or 14 day surface stay
7 day round trip
24 day free retrun abort - sized for worst case
** Volumes sized for 21 day nominal case

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

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Lunar Transfer/Excursion Crew Module
Direct Entry

35 day duration*
28 day surface stay

Crew Size	2	4	6	8
Habitable Volume (m3)	66	132	198	264
ECLSS	1,293	1,835	2,692	3,549
Structure	2,801	3,901	4,838	6,045
Command/Control/Power	1,703	2,476	3,232	4,021
Man-Systems	417	457	497	537
Consumables	893	1,718	2,578	3,436
Personnel and Effects, Spares, etc.	305	489	673	923
Radiation Shelter	778	1,202	1,626	2,050
Earth Entry Heat Shield	2,600	4,212	5,501	6,581
Earth Recovery Equipment	647	977	1,298	1,629
Mass Growth (15%)	1,376	2,190	2,878	3,581
Total Module Mass	12,813	19,457	25,813	32,352

* 28 day surface stay
7 day round trip time

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

Lunar Transfer/Excursion Crew Module
Direct Entry

49 day duration*
42 day surface stay

Crew Size	2	4	6	8
Habitable Volume (m3)	84	168	252	336
ECLSS	1,693	2,403	3,546	4,687
Structure	3,145	4,586	6,002	7,442
Command/Control/Power	1,703	2,476	3,232	4,021
Man-Systems	417	457	497	537
Consumables	1,146	2,289	3,435	4,579
Personnel and Effects, Spares, etc.	305	489	673	923
Radiation Shelter	778	1,202	1,626	2,050
Earth Entry Heat Shield	3,094	4,943	6,456	7,757
Earth Recovery Equipment	737	1,131	1,528	1,919
Mass Growth (15%)	1,673	2,511	3,359	4,181
Total Module Mass	**14,691**	**22,487**	**30,354**	**38,096**

* 42 day surface stay
7 day round trip time

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

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LEV Hab Modules

Assumptions

- STCAEM Crew Module Structural System
  - SSF diameter (4.4m) cylinder section
  - All penetrations occur in the cylinder section
  - All structural attachments occur at girth rings
  - Common ellipsoidal end domes (2:1 aspect ratio)
- Volume/crew consistent with historical spacecraft data
- Open ECLS system
- Power supply via fuel cells/solar arrays
  - 1 day = fuel cells
  - 14 + days = solar night capacity
- Minimum mass airlock - 1 airlock cycle/day per 2 crew
- No radiation shielding
- Human waste and urine storage
  - 1 day = bags and storage recepticles
  - 14 + days = toilet and storage bags
- Food - all shelf stable - 1.25 kg/man day
- Minimum medical provisions
- 15% mass growth

Operating Mode

- In conjunction with a transfer vehicle (LTV/LEV scenario)
- Excursion module only - no direct Earth entry or transfer

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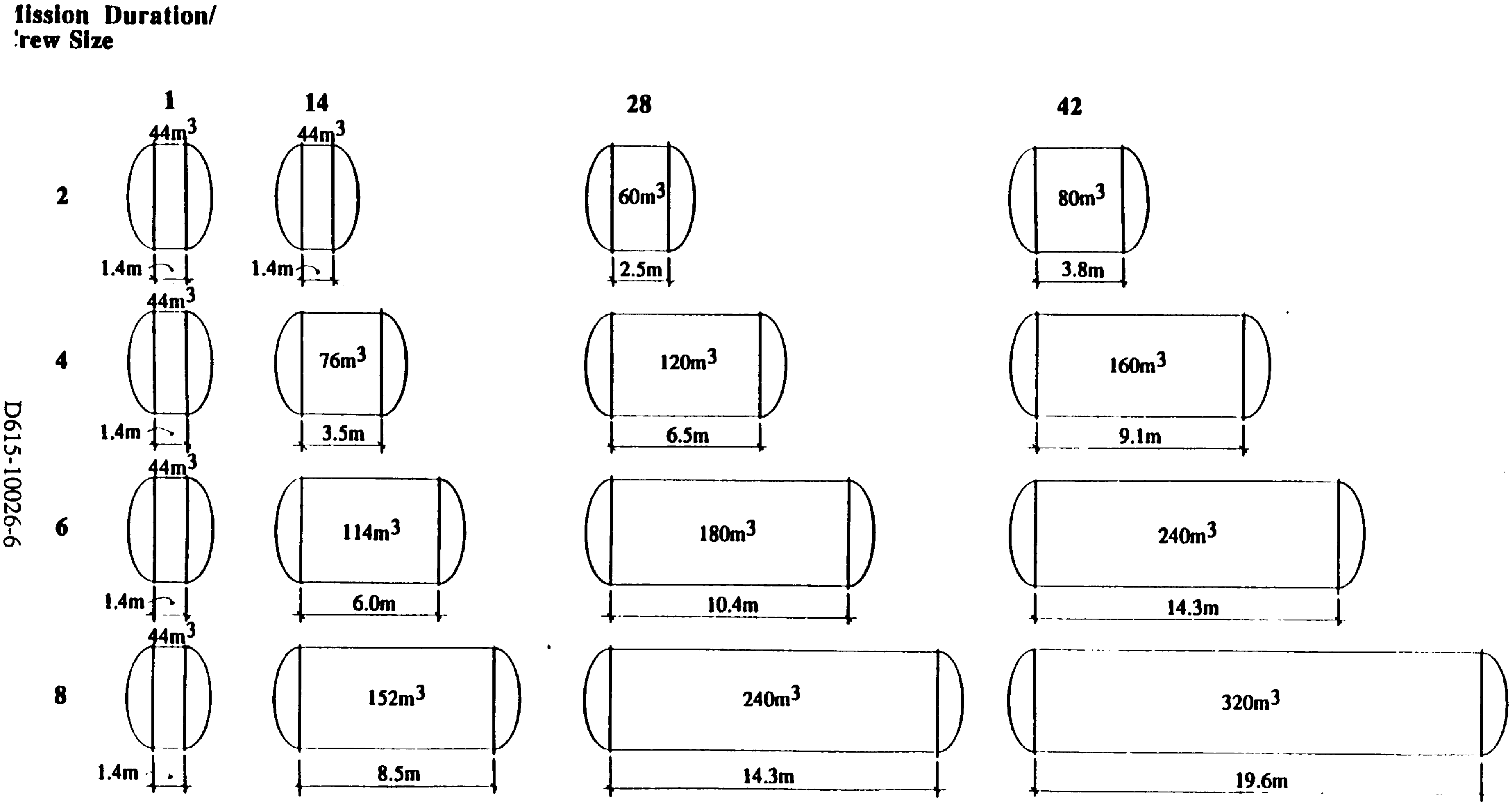
Lunar Excursion Modules Configuration Envelopes

This chart shows the relative sizes of excursion modules for crews of 2,4,6 and 8 and surface stays of 1, 14, 28 and 42 days.

Lunar Excursion Modules
Configuration Envelopes



NOTE: All modules 4.4m diameter

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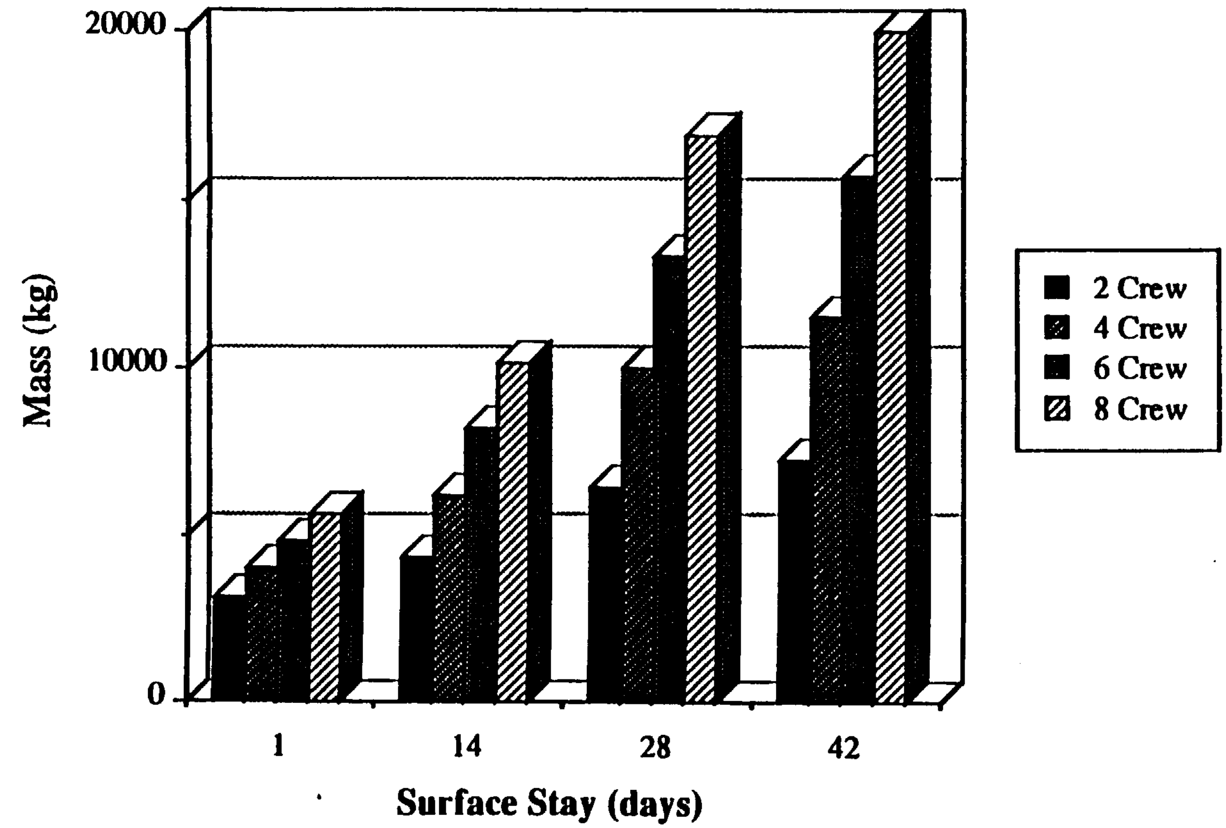
Lunar Excursion Module Mass Summary

This chart shows the relative masses of the excursion modules shown on the previous chart. As shown, the modules for 6 and 8 crew for 28 and 42 day surface stays become unusually large and massive.

Lunar Excursion Module Mass Summary



- Operates in conjunction with a transfer vehicle
- 1 day duration includes 1 repress
- 14 and over durations include an airlock and 2 represses

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Lunar Excursion Crew Module

1 day duration

Crew Size	2	4	6	8
Habitable volume (m3)	44	44	44	44
ECLSS Total	**228**	**341**	**454**	**566**
ARS/ACS/ACMA	11	21	32	42
1 Repress (incl. 40kg plumbing)	95	95	95	95
Temperature and Humidity Control	96	192	288	384
Thermal Control System	20	20	20	20
Potable Water Storage System	6	13	19	25
Structure Total	**1,630**	**1,665**	**1,700**	**1,735**
Primary/Secondary Structure	1,032	1,032	1,032	1,032
Berthing ring/mech. (1)	139	139	139	139
Berthing interface plate (1)	90	90	90	90
Hatches (2)	134	134	134	134
Windows	100	100	100	100
Chairs	10	20	30	40
Vents/Plumbing	125	150	175	200
Command/Control/Power Total	**323**	**409**	**495**	**580**
ECWS/DMS	100	100	100	100
Fault Detection & Isolation	---	---	---	---
Power System (Fuel cells, cond. eq., solar arrays)	173	259	345	430
Lights	50	50	50	50
Man System Total	**504**	**1,008**	**1,492**	**1,976**
WMS/Waste storage	4	8	12	16
Personnel and Effects	160	320	480	640
Transfer EVA Suits with PLSS	340	680	1,000	1,320
Consumables Total	**32**	**38**	**45**	**51**
Food and Packaging	3	5	8	10
Other consumables	4	8	12	16
Tools	25	25	25	25
Weight Growth Total (15%)	**408**	**519**	**628**	**736**
Total Module Mass	**3,125**	**3,980**	**4,814**	**5,644**

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

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Lunar Excursion Crew Module

14 day duration

Crew Size	2	4	6	8
Habitable volume (m3)	44	76	114	152
ECLSS Total	**561**	**971**	**1,397**	**1,822**
ARS/ACS/ACMA	147	294	441	588
2 Repress(100kg plumbing/pumps)	210	290	385	480
Temperature and Humidity Control	96	192	288	384
Thermal Control System	20	20	20	20
Potable Water Storage System	88	175	263	350
Structure Total	**2,140**	**2,623**	**3,194**	**3,766**
Primary/Secondary Structure	1,032	1,480	2,016	2,553
Airlock	510	510	510	510
Berthing ring/mech. (1)	139	139	139	139
Berthing interface plate (1)	90	90	90	90
Hatches (2)	134	134	134	134
Windows	100	100	100	100
Chairs	10	20	30	40
Vents/Plumbing	125	150	175	200
Command/Control/Power Total	**430**	**574**	**714**	**855**
ECWS/DMS	100	100	100	100
Power System	280	424	564	705
(Fuel cells, cond. eq., solar arrays)				
Lights	50	50	50	50
Man System Total	**556**	**1,112**	**1,648**	**2,184**
WMS/Waste storage	56	112	168	224
Personnel and Effects	160	320	480	640
Transfer EVA Suits with PLSS	340	680	1,000	1,320
Consumables Total	**64**	**103**	**142**	**181**
Food and Packaging	35	70	105	140
Other consumables	4	8	12	16
Tools	25	25	25	25
Weight Growth Total (15%)	**563**	**807**	**1,064**	**1,321**
Total Module Mass	**4,314**	**6,190**	**8,159**	**10,129**

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

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Lunar Excursion Crew Module

28 day duration

Crew Size	2	4	6	8
Habitable volume (m3)	60	120	180	240
ECLSS Total	**835**	**1,550**	**2,265**	**2,980**
ARS/ACS/ACMA	294	588	882	1,176
2 Repress(100kg plumbing/pumps)	250	400	550	700
Temperature and Humidity Control	96	192	288	384
Thermal Control System	20	20	20	20
Potable Water Storage System	175	350	525	700
Structure Total	**2,373**	**3,267**	**4,029**	**5,010**
Primary/Secondary Structure	1,265	2,124	2,851	3,797
Airlock	510	510	510	510
Berthing ring/mech. (1)	139	139	139	139
Berthing interface plate (1)	90	90	90	90
Hatches (2)	134	134	134	134
Windows	100	100	100	100
Chairs	10	20	30	40
Vents/Plumbing	125	150	175	200
Command/Control/Power Total	**1,703**	**2,476**	**3,232**	**4,021**
ECWS/DMS	100	100	100	100
Power System (Fuel cells, cond. eq., solar arrays)	1,553	2,326	3,082	3,871
Lights	50	50	50	50
Man System Total	**612**	**1,224**	**1,816**	**2,408**
WMS/Waste storage	112	224	336	448
Personnel and Effects	160	320	480	640
Transfer EVA Suits with PLSS	340	680	1,000	1,320
Consumables Total	**99**	**173**	**247**	**321**
Food and Packaging	70	140	210	280
Other consumables	4	8	12	16
Tools	25	25	25	25
Weight Growth Total (15%)	**843**	**1,304**	**1,738**	**2,211**
Total Module Mass	**6,465**	**9,994**	**13,327**	**16,951**

Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

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Lunar Excursion Crew Module

42 day duration

Crew Size	2	4	6	8
Habitable volume (m3)	80	160	240	320
ECLSS Total	**1,120**	**2,119**	**3,119**	**4,118**
ARS/ACS/ACMA	441	882	1,323	1,764
2 Repress(100kg plumbing/pumps)	300	500	700	900
Temperature and Humidity Control	96	192	288	384
Thermal Control System	20	20	20	20
Potable Water Storage System	263	525	788	1,050
Structure Total	**2,652**	**3,824**	**4,975**	**6,146**
Primary/Secondary Structure	1,544	2,681	3,797	4,933
Airlock	510	510	510	510
Berthing ring/mech. (1)	139	139	139	139
Berthing interface plate (1)	90	90	90	90
Hatches (2)	134	134	134	134
Windows	100	100	100	100
Chairs	10	20	30	40
Vents/Plumbing	125	150	175	200
Command/Control/Power Total	**1,703**	**2,476**	**3,232**	**4,021**
ECWS/DMS	100	100	100	100
Power System	1,553	2,326	3,082	3,871
(Fuel cells, cond. eq., solar arrays)				
Lights	50	50	50	50
Man System Total	**668**	**1,336**	**1,984**	**2,632**
WMS/Waste storage	168	336	504	672
Personnel and Effects	160	320	480	640
Transfer EVA Suits with PLSS	340	680	1,000	1,320
Consumables Total	**134**	**243**	**352**	**461**
Food and Packaging	105	210	315	420
Other consumables	4	8	12	16
Tools	25	25	25	25
Weight Growth Total (15%)	**942**	**1,500**	**2,049**	**2,607**
Total Module Mass	**7,219**	**11,498**	**15,711**	**19,985**

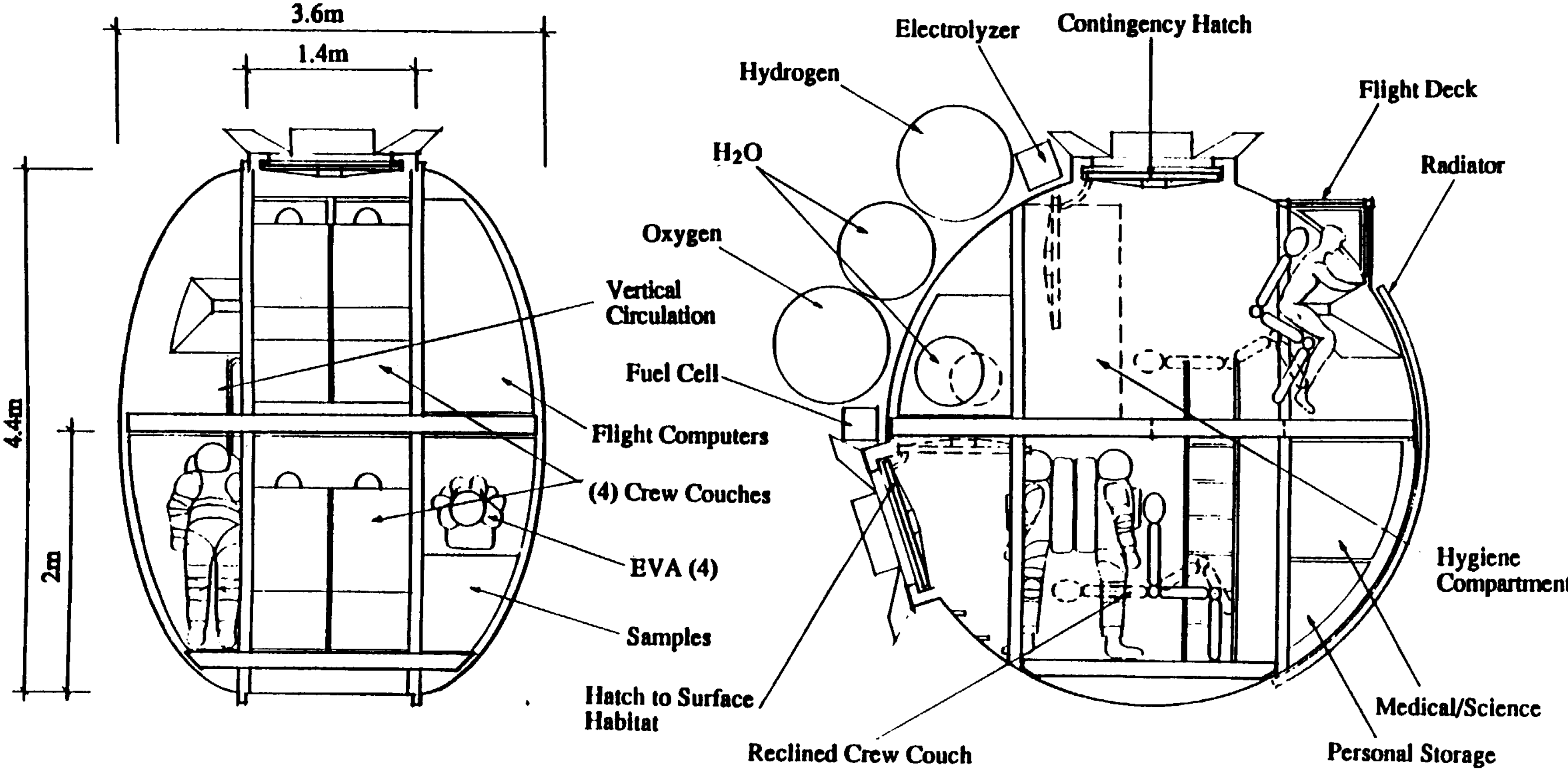
Note: All weights are in kilograms
Lunar sample material mass not included

PRELIMINARY

Common Short-Duration Crew Module

The next 2 charts show a detailed interior configuration of and excursion module that can accommodate 2 crew for a 1 or 14 day surface stay, and 4, 6 and 8 crew for a one day surface stay.

Common Short-Duration Crew Module



Cross Section

Long Section

- SSF diameter cylinder.
- All penetrations occur in cylinder section.
- All structural attachments at girth rings.
- Common ellipsoidal end domes.
- 43.6m3 total volume

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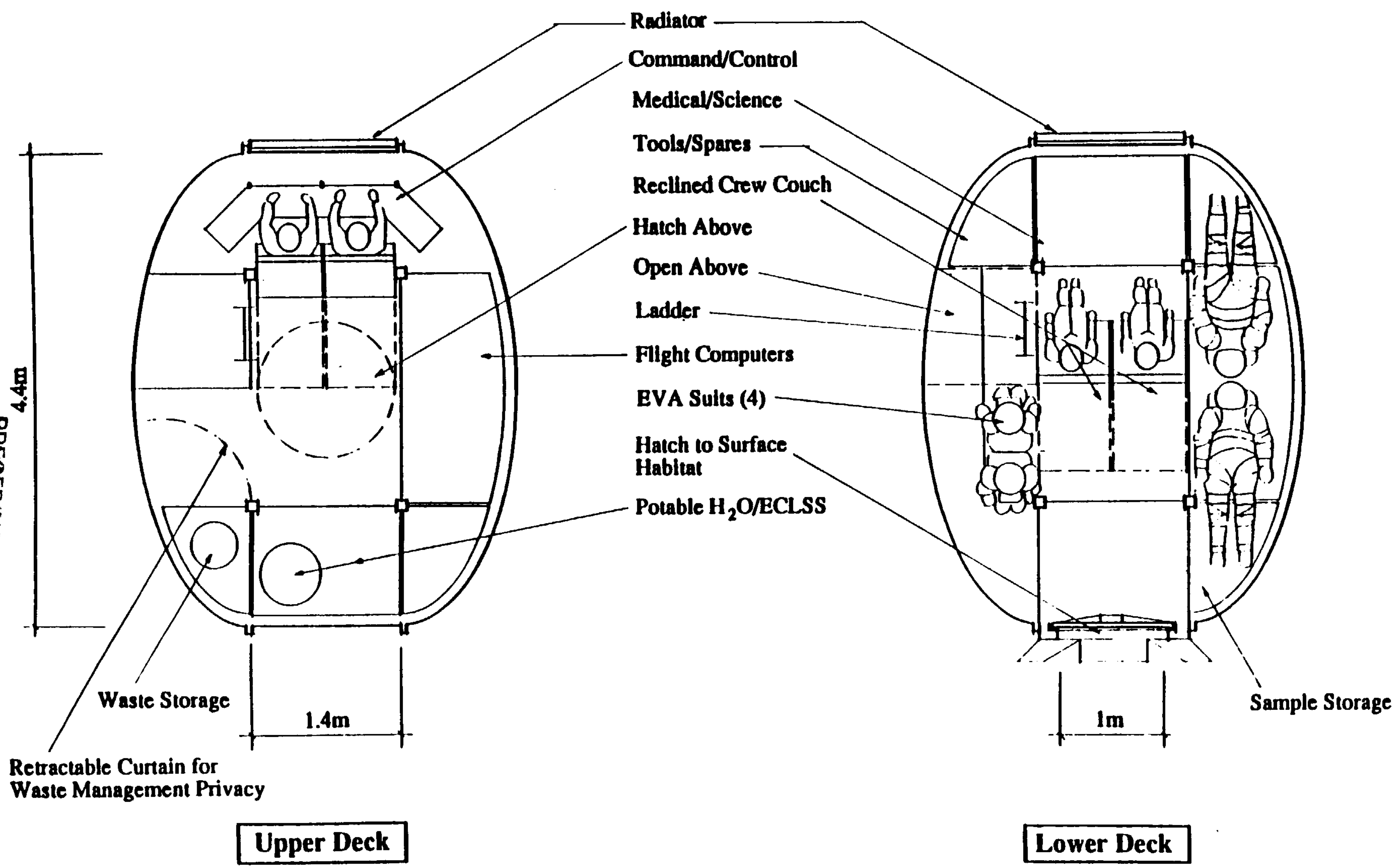
Common Short-Duration Crew Module



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Excursion Vehicle Crew Cab Duration Trade

This trade was conducted in response to a request from Level II, to determine the staytime after which it is lighter to provide a separate surface habitation module which remains behind, than to embed this function in the excursion vehicle crew cab, which ascends back to orbit. The operating mode for this study is an excursion scenario (including "campsite" options), with no permanent base available. The chart outlines the assumptions used in the analysis.

ADVANCED CIVIL SPACE SYSTEMS

EV Crew Cab Duration Trade

BOEING

Level II Question

Which is lighter:
- Sizing the EV for the entire surface duration
- Using a 1 d EV cab along with a separate surface habitat

Assumptions

General

- Crew sizes of 2, 4, 6, and 8
- 2 kg food/person/day; 6 kg water
- Open ECLSS, 3 repressurizations
- 2,000 kg of science equipment, left on the surface; 500 kg return payload
- Propellant tanks sized for 25 t load except when > 25 t is needed
- 5% fuel contingency, 2% boiloff

Lunar

- In cab-plus-hab case, ascent vehicle returns to LLO without landing legs and some structure
- In cab-only case, entire vehicle returns to LLO for reuse in LOR mode

Mars

- Separate ascent and descent stages for all cases
- 20 % aerobrake for entry, jettisoned prior to landing

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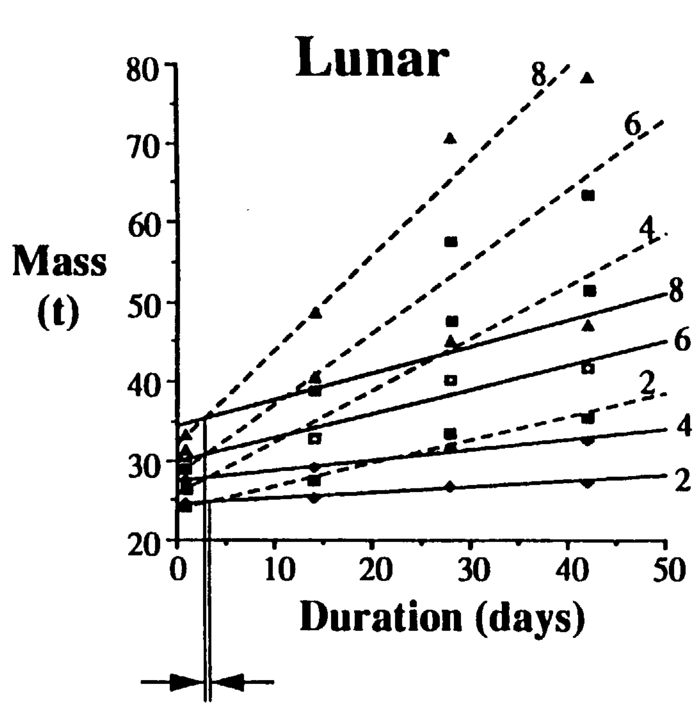
EV Cab Duration Trade Data

The graphed data reveal that for anything longer than a few days' duration, launching the habitation system required for that duration back to orbit is the heavier option, for both the Moon and Mars.

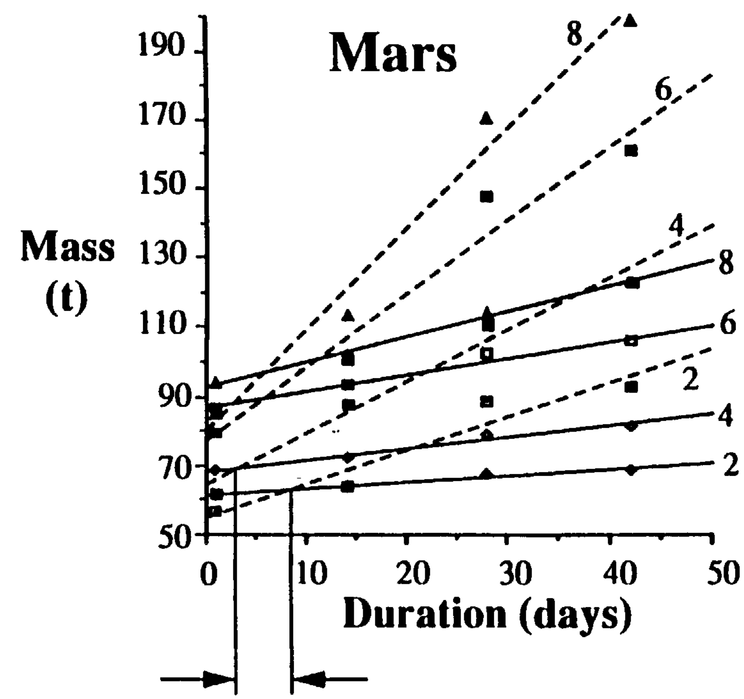
The continued slope of the curves beyond the crossover region is an artifact of the analytical assumption of an open ECLSS system, and therefore these curves should not be used as design parametrics. In fact, the mass-crossover to ECLSS closure would show up as a knee in the curves between 10 - 20 days, resulting in lower slope.

EV Cab Duration Trade Data

Curves parametrized by crew size
——— 1 d EV cab & surface module ‑‑‑‑‑‑‑‑ EV cab for duration



3 d crossover



3 - 8 d crossover

A separate habitation module is the preferred solution for both lunar and Mars surface stays exceeding a few days

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C. ACRV Modifications

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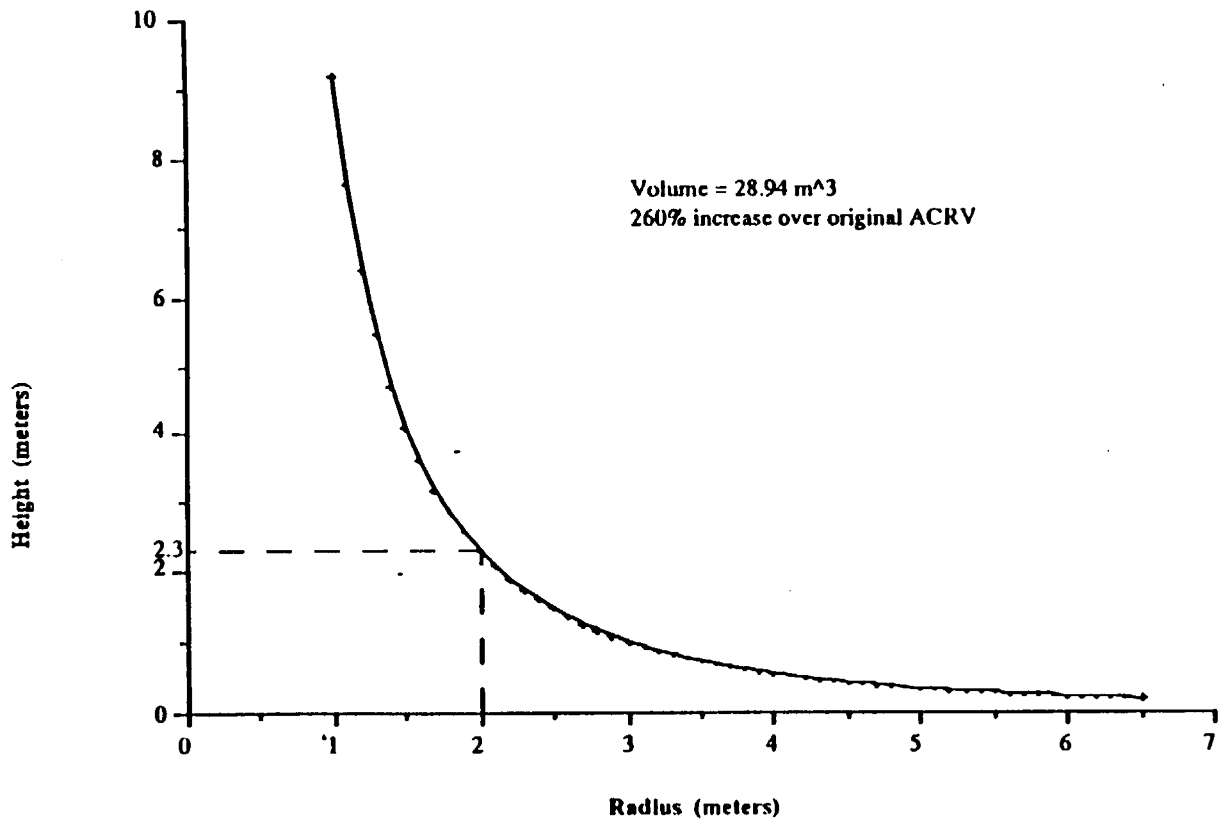
ACRV MODIFICATIONS

ASSUMPTIONS:

Initial Mass	5020 kg
Initial Volume	11.13 m^3
Apollo Living Space Percentage	46.6%
Maximum Trip Time	15 days
Consumable Masses	
Water and Tanks	350 kg
Food and Packaging	120 kg
O2/LiOH and HP Tank	200 kg
O2 and Cryo Tank	60 kg
Consumable Densities	
Food	500 kg/m^3
Water, Oxygen	1100 kg/m^3
Consumable Volume	1 m^3
EVA Equipment Volume	0.75 m^3/person
Return Sample Volume	1.5 m^3

Volume Resizing for the ACRV



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ACRV MODIFICATIONS

ASSUMPTIONS:

* Fraction of free volume to total volume is the same as for Apollo (46.6%)
* Free Volume (FV) = Space available not taken by built-in equipment
* Crew Volume (CV) = Space available for crew habitation
  CV (total) = FV - Consumable Volume - EVA Volume - Sample Volume
* Sample Volume is based on the Lunar Sample Return Containers as described by the JPL's <u>SEI Science Payloads</u> document.
* Individual Crew Space on the order of Apollo (approximately 2 m^3 / person).
* Original mass statement: <u>Assured Crew Return Vehicle, AF(6)1-S1(6), Mass Properties Status Report No. 2</u>, 27 April 1990.
* Modifications based upon trip duration, required equipment and systems, and consultation with those who know better.
* Service Module may be employed to transport the necessary tankage, propellant, oxygen, etc. (no estimate of the mass has been made).
* Assume that the ACRV carries enough extra oxygen, etc., for multiple EVAs on the Lunar Surface based upon over-estimation and contingency.
* On-board storage: 1 m^3 for consumables, 1.5 m^3 for return payload, and 0.75 m^3 per EVA suit.
* Consumable Mass is not accounted for in the ACRV mass. There is the nominal volume on-board consumable storage.

FINDINGS:

Volume Increase:	260%
New Volume:	28.94 m^3
Free Volume:	13.49 m^3
Diameter:	4.0 m
Height:	2.3 m
Crew Space:	
Total	7.99 m^3
Individual	2.00 m^3
Mass:	7.4 metric tons

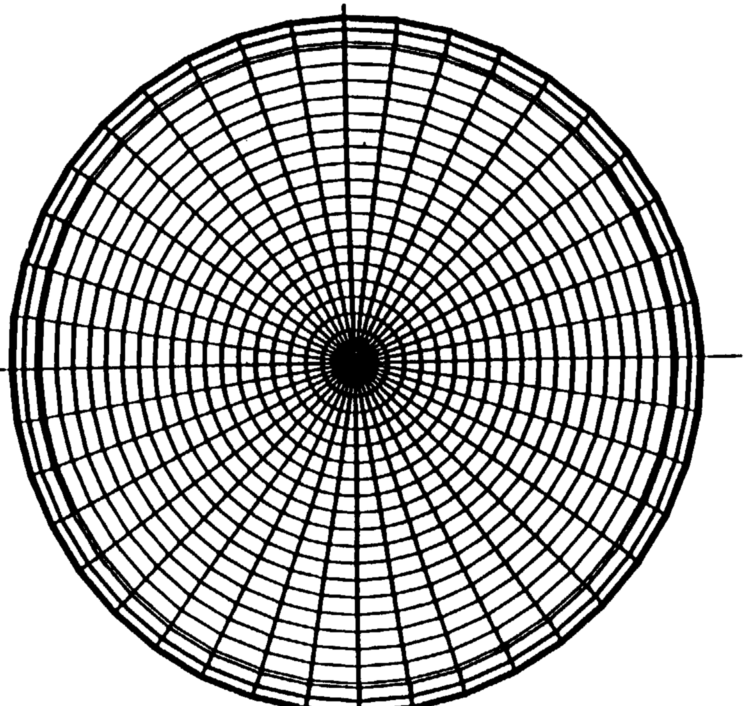
D. Aerobrake

Lunar Transportation Family
Aerobrake

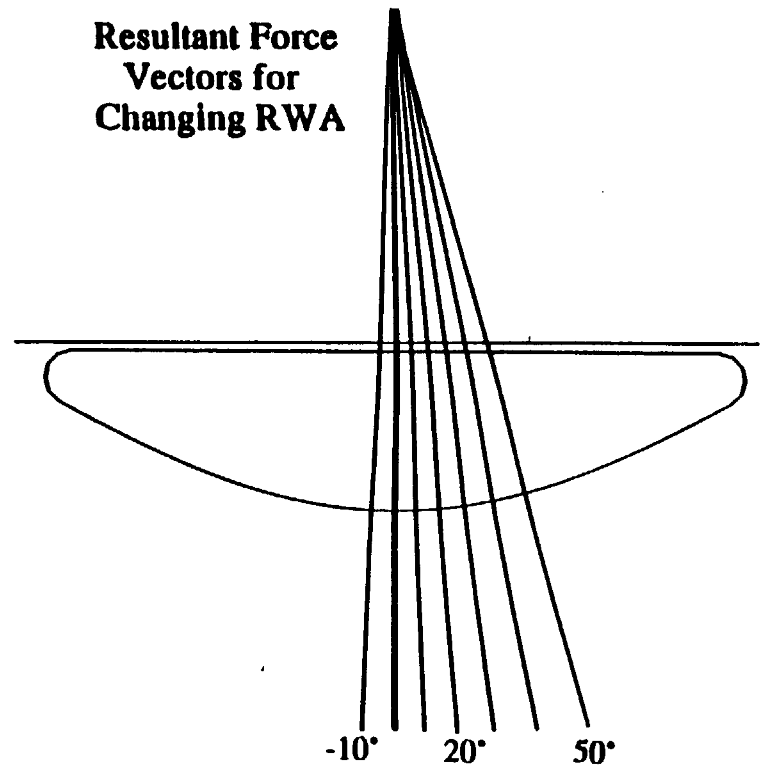
The aerobrake shape shown subsequently is a symmetrical hyperboloid ~ 20 m in diameter. The shape is derived from the STCAEM L/D = 0.5 aerobrake so that it may be possible to evolve from a lunar to a Mars aerobrake without 2 totally different hardware development programs.

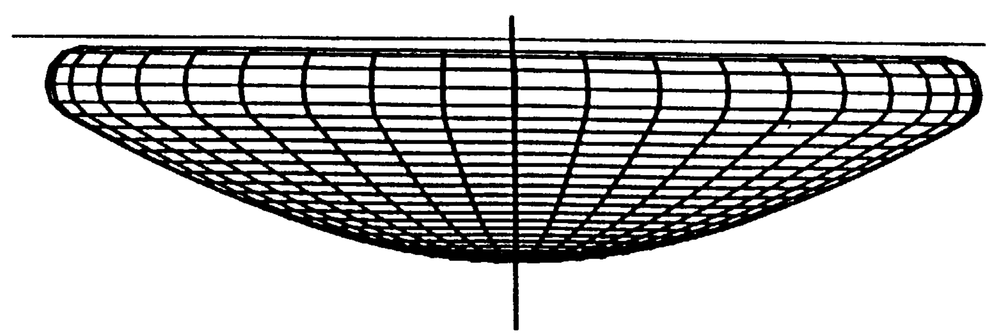
Values for Shape Parameters	
Semimajor Axis Ratio	1.2000
Eccen. of Body of Revol.	1.6000
Eccen. of Cutting Cyl.	0.0060
Truncation/SMA Ratio	0.0000
Lip radius/SMA	0.0800
Lip taper ratio	1.0000
Plan Area	4.8856
Surface Area	6.8070

Angle	CL	CD	L/D	Moment arm
-10	-0.1930	1.4669	-0.1316	0.0618
0	0.0044	1.5170	0.0029	0.0059
10	0.2009	1.4624	0.1374	-0.0501
20	0.3611	1.3130	0.2750	-0.1083
30	0.4564	1.0958	0.4165	-0.1711
40	0.4764	0.8483	0.5615	-0.2410
50	0.4298	0.6086	0.7061	-0.3197



Resultant Force Vectors for Changing RWA





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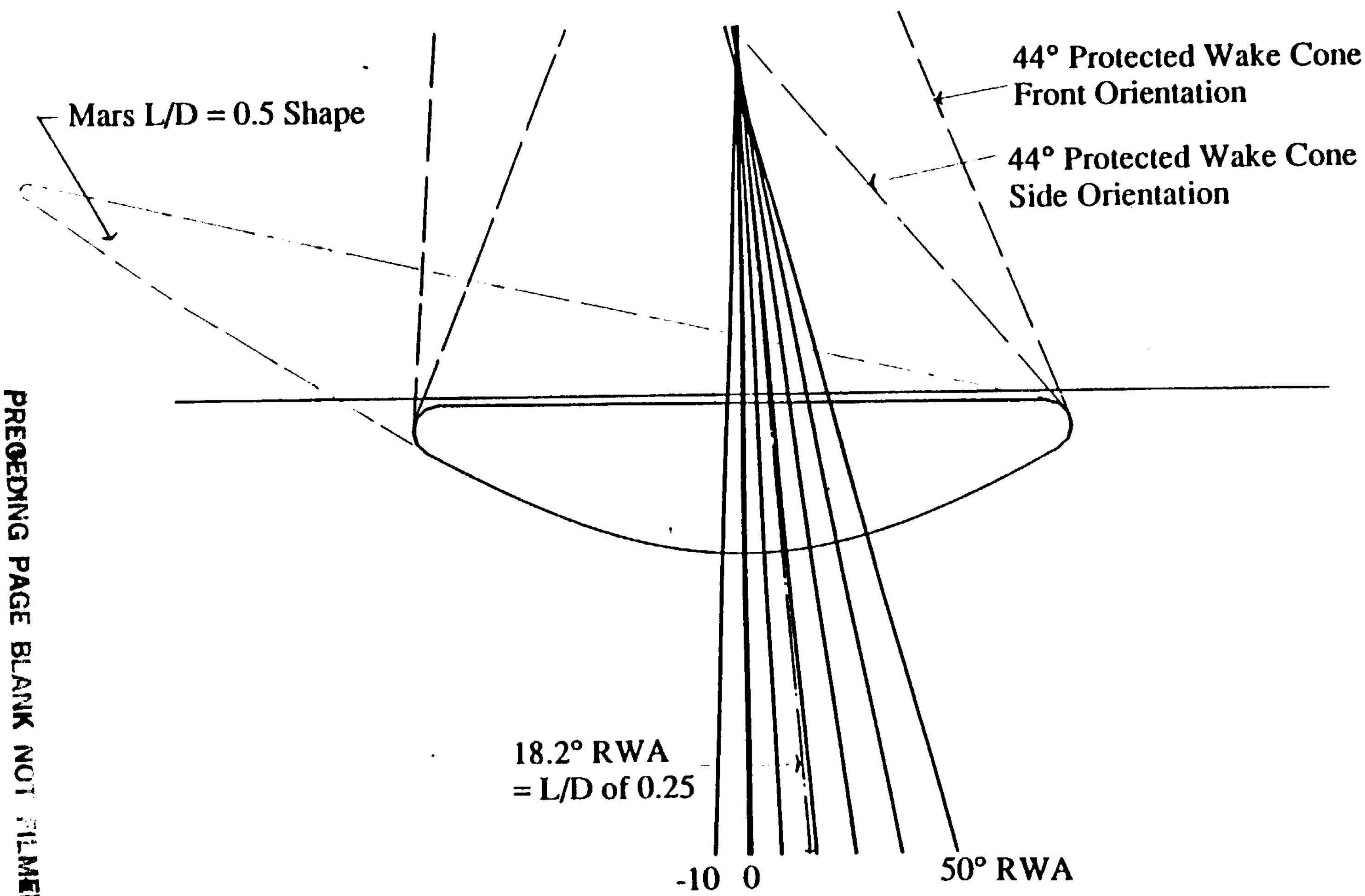
Lunar Transportation Family

Aerobrake



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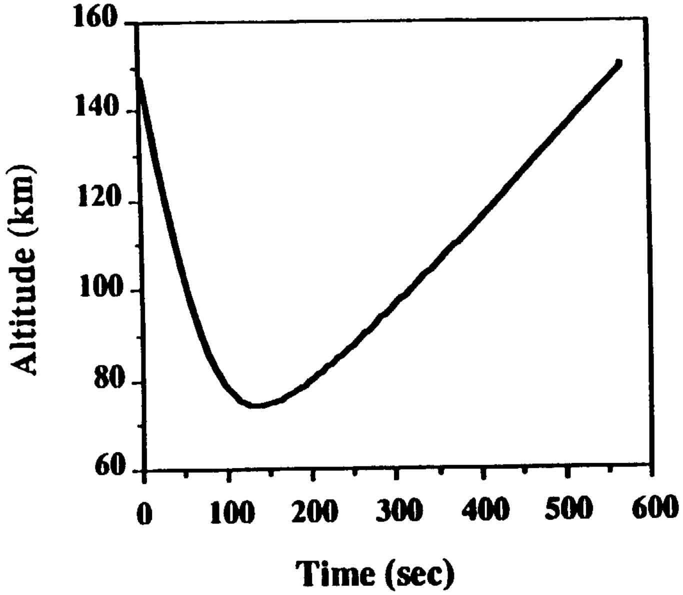
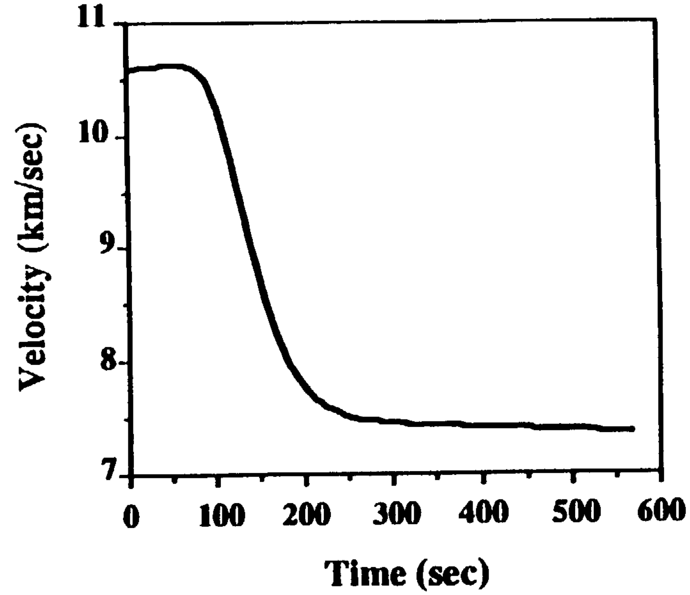
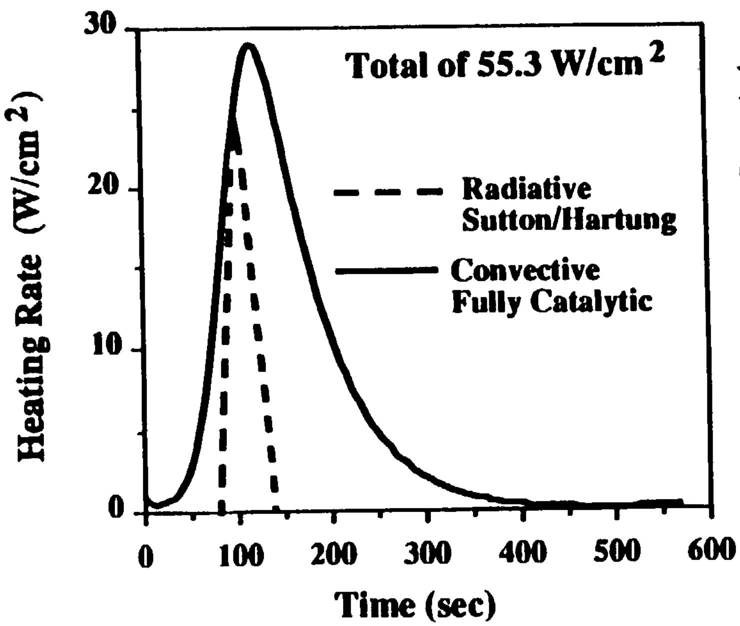
LTV Trajectory and Stagnation Point Heating

For the worst case heating, the trajectory was simulated using a L/D = 0.0 (ballistic trajectory), which is the middle of the aerocapture corridor. The LTV Earth return mass was 26.5 metric tons. The aerobrake for the LTV has an L/D capability of 0.25, with a diameter of 22 meters and reference area of 380m^2. The ballistic coefficient of this vehicle is 52.1 kg/m^2. An effective nose radius of 13 meters was used for the stagnation point heating. Shown below are the resulting trajectory and the stagnation point convective and radiative heating rates along these trajectories. Radiative heating rates are based on the Sutton and Hartung tabular values. Reduced heating rates for the LTV can be accomplished with down lift trajectories allowing the vehicle to deccelerate higher up in the atmosphere were the density is smaller.

LTV Trajectory and Stagnation Point Heating







Nose Radius 13 meters

Target Apoapsis 407 km

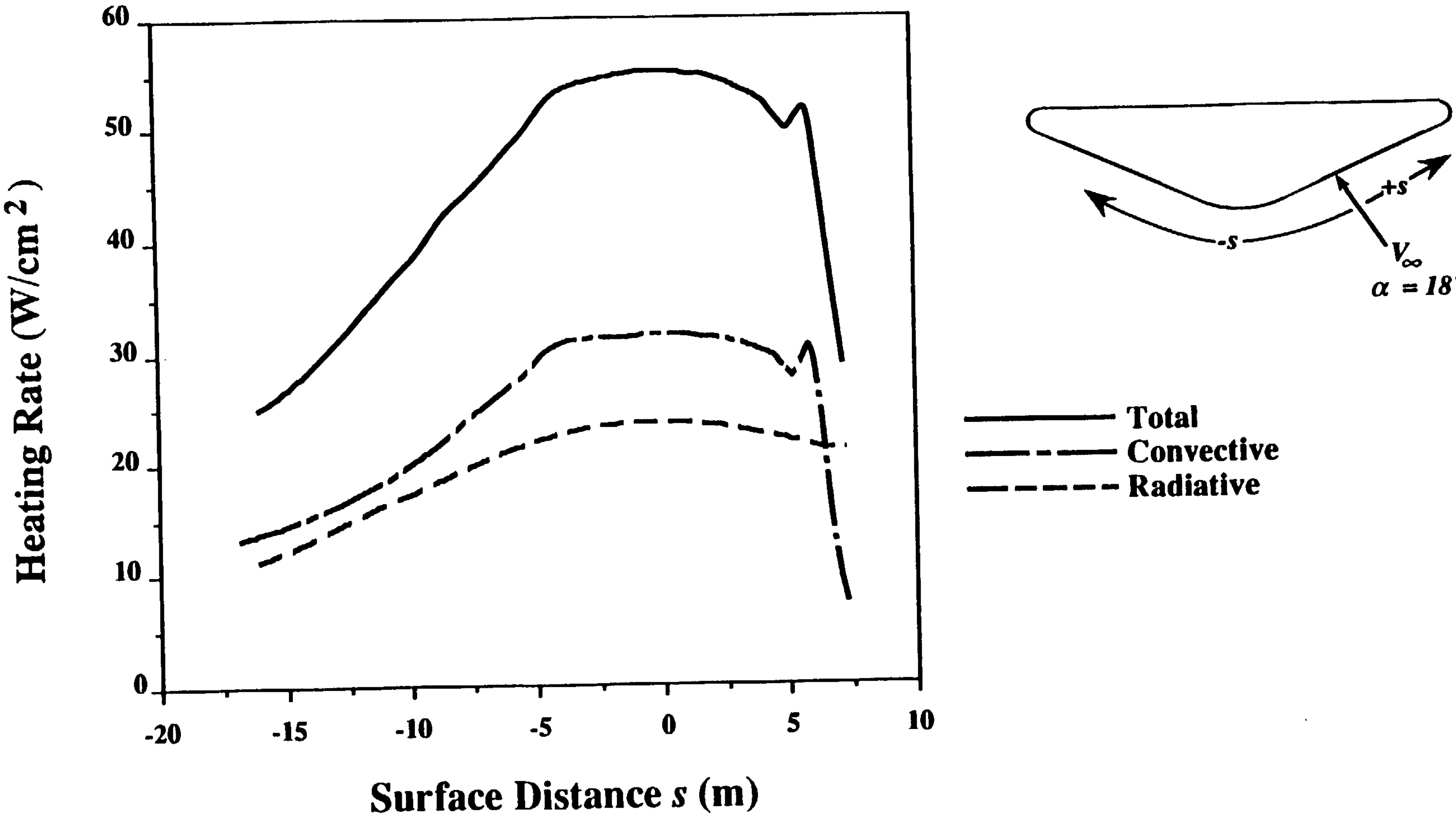
Entry Altitude 150 km

Entry Velocity 11.05 km/sec

LTV Centerline Heating

The chart below displays the centerline heating for the LTV aerobrake, calculated at the maximum stagnation point heating along the previously presented trajectory. This maximum heating occured at 98 seconds into the capture trajectory at an altitude of 78.8 km and a velocity of 10.32 km/sec. The maximum total stagnation point heating was 55.3 W/cm^2 . Convective heating rates were calculated using the Boeing Boundary Layer Analysis Program with Peng and Pindroh transport properties for air (equilibrium). Pressure distributions for the aerobrake were based on Modified Newtonian Impact Theory, and unswept cylinder theory for the lip of the brake. The radiative heating rates were calculate using the tabular values of Sutton and Hartung (equilibrium).

LTV Centerline Heating



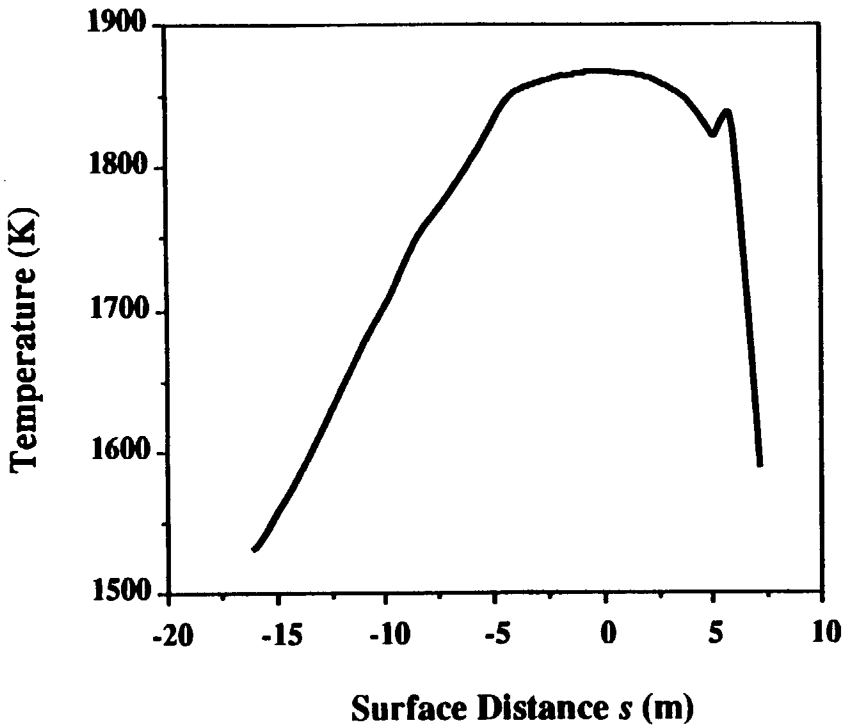
LTV Equilibrium Wall Temperatures

Equilibrium wall temperatures along the centerline and side streamline of the LTV aerobrake are shown below. These temperatures were computed, with an emissivity of 0.8, using the Stefan-Botlzmann relationship. The side streamline heating rates were calculated as shown on the previous chart. The maximum wall temperature is 1870 K.

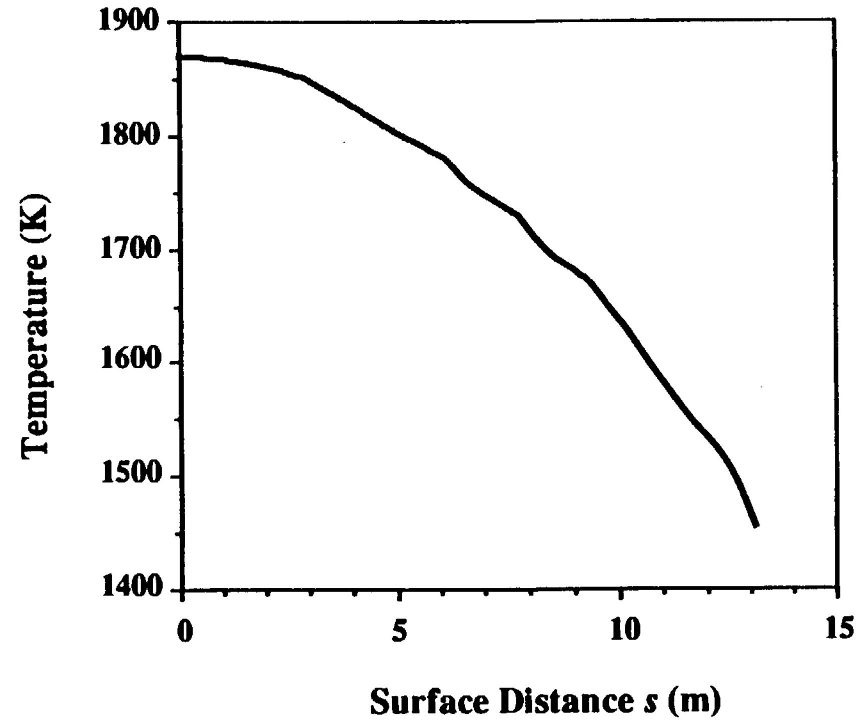
LTV Equilibrium Wall Temperatures

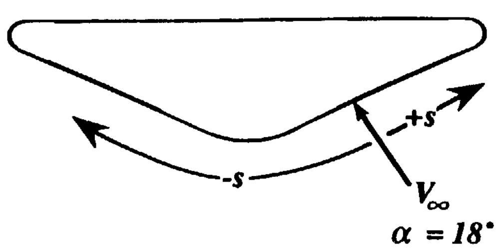
Emissivity = 0.8

Centerline

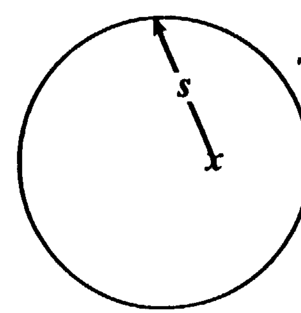


Side Streamline





Side View



Top View

LTV Temperature Distribution

The following chart displays the temperature contours for the LTV aerobrake based on the streamline temperatures (both the centerline and side line streamlines were used). The LTV aerobrake will require advanced reradiatives, if it is to be reusable, due to the above 1800 K temperatures.

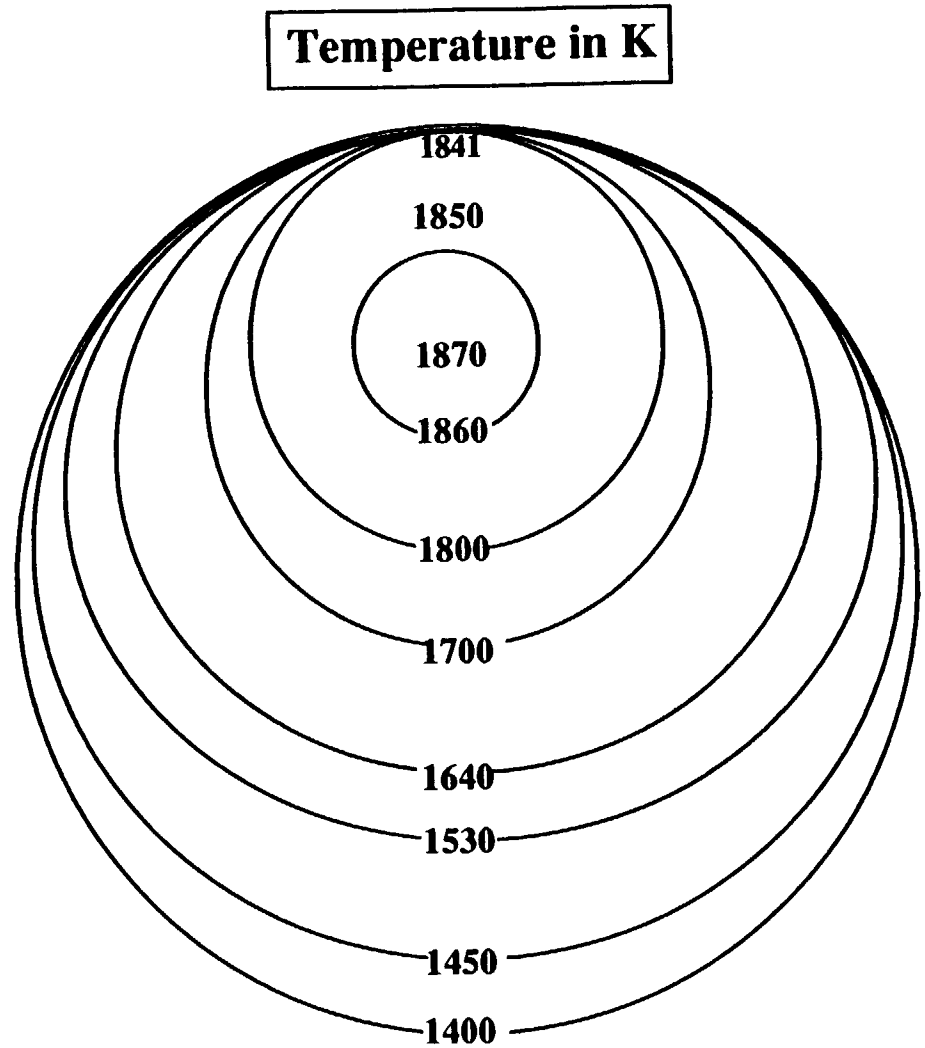
LTV Temperature Distribution

Assumptions

- Radiative - Sutton/Hartung
- Convective - BLAPP
- Emissivity = 0.8

Conditions at maximum heating

- Velocity : 10.32 km/sec
- Altitude : 78.75 km
- Radiative : 23.7 W/cm^2
- Convective : 31.6 W/cm^2



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E. Cost

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Programmatics

The objectives of the Programmatics task during the current phase of the study were: (1) realistic initial schedules that include initial critical path program elements; (2) initial descriptions of new or unique facilities requirements; (3) development of a stable, clear, responsive work breakdown structure (WBS) and WBS dictionary; (4) initial realistic estimates of vehicle, mission and program costs, cost uncertainties, and funding profile requirements; (5) initial risk analysis, and (6) early and continuing infusion of programmatics data into other study tasks to drive requirements/design/trade decisions.

The issues addressed during the study to date included: (1) capturing all potential long-lead program items such as precursor missions, technology advancement and advanced development, related infrastructure development, support systems and new or modified facility construction, since these are as important as cost and funding in assessing goal achievability; (2) incorporating sufficient operating margin in schedules to obtain high probability of making the relatively brief Mars launch windows; (3) the work breakdown structure must support key study goals such as commonality and (4) cost estimating accuracy and uncertainty are recurring issues in concept definition studies.

Introduction

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown in the "Overall Study Flow" chart. After the "neckdown" was completed, significant effort was put into programmatics.

As was indicated earlier, we established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for Lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in space transportation technologies as baselines for greater activity levels. The high level schedules developed for these three levels of activity are shown in the "Minimum Program", "Full Science Program" and " Industrialization and Settlement Program" charts and a comparison of them for both Lunar and Mars is shown in the "Lunar Program Comparison" and "Mars Program Comparison" charts.

Schedule/Network Development Methodology

A PC system called Open Plan by WST Corporation was used, which allows direct control and lower cost over a larger (mainframe) system. The network was purposely kept simple. Summary activities were used in development of the networks. When detailed to a lower level, some activities will require a different calendar than we used. One calendar with a five day work week - no holiday was used. Utilizing multicalendars on a summary network could confuse the development. The Preliminary WBS Structure Level 7 was followed for selection of work to be detailed. An example of Level 7 is: MEV Ascent Vehicle Structure/Mechanisms. We then developed a generic logic string of activities with standard durations for like activities. This logic was then applied against each WBS Level 7 element. To establish interface ties between logic strings and determination of major events, we used the Upper Level Summary Schedule and Summary Level Technology Schedule.

Goals/Purpose

There were two goals for the schedule/network development. These were:

a. Guidelines for Future Development. The schedules are a preliminary road map to follow in the development program.

b. Layout Basis Framework for Network. The networks can be used for future detail network development. This development can be in phases retaining unattended logic for areas which can be be detailed.

Status

Six preliminary networks have been developed. They are:

- Lunar minimum
- Lunar full science
- Lunar industrialization
- Mars missions
- Mars full science
- Mars settlement

These networks will be further developed as information becomes available The technology development plan schedules are shown in the Schedules section of this text; an example of the standard 6 year program phase C/D schedule is shown in the "Reference 6 yr. Full Scale Development Schedule" chart . The network schedules developed during the study are available in the Final Report Cost Data Book.

Facilities

The facility requirements and approaches are discussed in the Facilities section of this text.

Work Breakdown Structure

The approach to developing a WBS tree and dictionary was to use the Space Station Freedom Work Package One WBS as a point of departure to capture commonality, modularity and evolution potentials. We worked with MSFC to evolve the WBS illustrated in the six WBS charts given in this section. The WBS dictionary details are provided with the WBS tree in a separate deliverable document.

Cost Data

Overall Approach

Space transfer concept cost estimates were developed through parametric and detail estimating techniques using program/scenario plans and hardware and software descriptions combined with NASA and subcontractor data. Our estimating approach simulates the aerospace development and production environment. It also reflects program options not typical of aerospace programs. This flexibility allows assessment of innovative program planning concepts.

Several tools were employed in this analysis. For developing estimates the Boeing Parametric Cost Model (PCM) designed specifically for advanced system estimating was used. It utilizes a company-wide, uniform computerized data base containing historical data compiled since 1969. The second major tool is a Boeing developed Life Cycle Cost Model. The third tool is the Boeing developed Return on Investment (ROI) Analyses.

The approach to cost estimating was to use the PCM to establish DDT&E and manufacturing cost of major hardware components or to use other estimates, (e.g. Nuclear Working Group estimator) if they were considered superior and then feed them to the LCC model. Variations on equipment hardware or mission alternatives can be run through the LCC and then compared for a return on investment. This flow is illustrated in the "Costing Methodology Flow" charts. We were able to investigate alternative concepts quickly, giving system designers more data for evolving scenario/mission responsive concepts. Transportation concepts, trade studies, and "neckdown" efforts were supported by this approach.

Parametric Cost Model

PCM develops cost from the subsystem level and builds upward to obtain total program cost. Costs are estimated from physical hardware descriptions (e.g., weights and complexities) and program parameters (e.g., quantities, learning curves, and integration levels). Known costs are input directly into the estimate when available; the model assesses the necessary system engineering and system test efforts needed for integration into the program. The PCM working unit is man-hours, which allows relationships that tie physical hardware descriptions first to design engineering or basic factory labor, and then through the organizational structure to pick up functional areas such as systems engineering, test, and development shop. Using man-hours instead of dollars for estimating relationships enables more reliable estimates. The PCM features, main inputs, and results are shown in the "Boeing Parametric Cost Model (PCM)" chart. The applicable PCM results, in constant 1990 dollars, are then put into the Life Cycle Cost Model to obtain cost spreads for the various missions/programs. The various hardware components costed for the three different missions/programs are shown in the "LCCM Hardware Assignments" chart.

The development of space hardware and components needed to accomplish the three different Lunar/Mars missions were identified. These components are grouped into three different categories defined below.

HLLV(Heavy Lift Launch Vehicle) is the booster required to lift personnel, cargo and fuels into LEO and support the LEO node operations.

Propulsion Includes the space propulsion system required to transfer people, cargo and equipment out of LEO and into space. Space means Lunar, Mars and Earth destinations. Propulsion Systems also include an all-propulsive cryogenic Trans Mars Injection System (TMIS) for the Minimum Mission, the Nuclear Electric Propulsion Stage for the Settlement/Industrial Missions.

Modules Include the space systems that are required to transfer people, cargo and equipment from LEO to Lunar and Mars orbit; to de-orbit and sustain life and operations on the Lunar and Mars Surface; and, finally, to return personnel and equipment to LEO.

Cost Buildups

The PCM cost Model can be used directly to obtain complete DDT&E cost, including production of major test articles, by entering into the manufacturing section the equivalent numbers of units for each item, including the first flight article. However, when operated in this way, PCM does not give the first unit cost. To save time, we operated PCM so as to give first unit cost, which we needed for life cycle cost analyses, and used the first unit cost to manually estimate the test hardware content of the DDT&E program. The "wrap factors" shown in the cost buildup sheets were derived from the PCM runs as the factor that is applied to design engineering cost to obtain complete design and development costs, e.g. including non-recurring items such as systems engineering and tooling development.

Life Cycle Cost Model

The LCCM cost data is a composite of HLLV costs, launch base facilities cost estimate based on $/sq. ft. and parametric estimates derived from the Parametric Cost Model. The principal source of information is from the PCM. All hardware cost estimates, with the exception of HLLV, have been developed with this model.

The LCCM consists of three individual models. One model is for the Minimum Program Scale; the second is for the Full Science Program Scale; while the third model is for the Settlement/Industrialization Program Scale. The Minimum Program meets the President's Space Exploration Initiative (SEI) objectives. These capabilities include permanent Lunar facilities but not permanent human presence and three missions to Mars. The Full Science program not only meets the President's SEI objectives but also provides for long term bases for far-ranging surface exploration. The Settlement/Industrialization program accomplishes the objectives of the Minimum and Full Science program scales and additionally returns practical benefits to Earth. These models were developed using the three architecture levels described in the Boeing manifest worksheets. Total cost for each system are tabulated by year and each year's totals feed into a summary sheet that calculates the total program cost for each level. Since the LCCM results are mission related, not just vehicle related, they are not provided here but are available in the Final Report Cost Data Book. The LCCM was developed using Microsoft Excel version 2.2 for the Macintosh computer. Any Macintosh equipped with Excel 2.2 can be used to execute the model.

Return On Investment

One of the principal uses of the LCCM is to develop trades and return on investment for technology options. As shown in the "Costing Methodology Flow" chart , two separate life cycle cost models (which include DDT&E and production cost data derived from the parametric cost models) must be developed for each ROI case; a reference, and a case utilizing a technology option. The two life cycle cost streams are separately entered, and the ROI model is executed. The flow also illustrates that not all of the data entered into the life cycle cost model is derived from available costing software. Technical analysis must accompany this data. For example, the number of units which must be produced for the DDT&E program must be determined. This is done at the subsystem level based on knowledge of past programs, and proposed system/subsystem tests. Since the ROI analysis is mission related, not just vehicle related, the data is not presented here but is available in the Final Report Cost Data Book.

Results

A summary of the cost data produced by the PCM for the lunar family of vehicles are given in the PCM summaries included in this section. The PCM program was used to produce DDT&E and production cost estimates for each of our reference Mars and lunar vehicles to the subsystem level. The DDT&E costs generated by the PCM do not include all of the necessary hardware for the first mission vehicle. Hence all necessary additional units (prototypes,test units, lab units, etc.) were added into the vehicle cost buildups as shown in the "Lunar Cost Buildup" charts. The total DDT&E includes additional costs (e.g.. additional units in the DDT&E program), contractor fees and the engineering wrap factor. The total DDT&E from the cost buildup and the unit cost from the PCM are the primary vehicle cost inputs to the LCC model

Risk Analyses

Risk analyses were conducted to develop an initial risk assessment for the various architectures. This presentation of risk analysis results considers development risk, man-rating requirements, and several aspects of mission and operations risk.

Development Risk

All of the architectures and technologies investigated in this study incur some degree of development risk; none are comprised entirely of fully developed technology. Development risks are correlated directly with technological uncertainties. We identified the following principal risks:

Cryogenics - High-performance insulation systems involve a great many layers of multi-layer insulation (MLI), and one or more vapor-cooled shields. Analyses and experiments have indicated the efficacy of these, but demonstration that such insulation systems can be fabricated at light weight, capable of surviving launch g and acoustics loads, remains to be accomplished. In addition, there are issues associated with propellant transfer and zero-g gauging. These, however, can be avoided for early lunar systems by proper choice of configuration and operations, e.g. the tandem-direct system recommended elsewhere in this report. This presents the opportunity to evolve these technologies with operations of initial flight systems.

Engines - There is little risk of being able to provide some sort of cryogenic engine for lunar and Mars missions. The RL- 10 could be modified to serve with little risk; deep throttling of this engine has already been demonstrated on the test stand. The risk of developing more advanced engines is also minimal. An advanced development program in this area serves mainly to reduce development cost by pioneering the critical features prior to full-scale development.

Aerocapture and aerobraking - There are six potential functions, given here in approximate ascending order of development risk: aero descent and landing of crew capsules returning from the Moon, aerocapture to low Earth orbit of returning reusable lunar vehicles, landing of Mars excursion vehicles from Mars orbit, aero descent and landing of crew capsules returning from Mars, aerocapture to low Earth orbit of returning Mars vehicles, and aerocapture to Mars orbit of Mars excursion and Mars transfer vehicles. The "Development Risk Assessment for Aerobraking by Function" chart provides a qualitative development risk comparison for these six functions.

Aerocapture of vehicles requires large aerobrakes. For these to be efficient, low mass per unit area is required, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials. By comparison, the crew capsules benefit much less from high performance structures and TPS.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk that has not yet been solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both. While the design challenge can probably be met, aerobrake assembly is a difficult design and development challenge, representing an important area of risk.

Nuclear thermal rockets - The basic technology of nuclear thermal rockets was developed and demonstrated during the 1960s and early 1970s. The development risk to reproduce this technology is minimal, except in testing as described below. Current studies are recommending advances in engine performance, both in specific impulse (higher reactor temperature) and in thrust-to-weight ratio (higher reactor power density). The risks in achieving these are modest inasmuch as performance targets can be adjusted to technology performance.

Reactor and engine tests during the 1960s jetted hot, slightly radioactive hydrogen directly into the atmosphere. Stricter environmental controls since that time prohibit discharge of nuclear engine effluent into the atmosphere. Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines. Full- containment facilities will be required to contain all the hydrogen effluent, presumably oxidize it to water, and remove the radioactivity.

Electric Propulsion Power Management and Thrusters - Power management and thrusters are common to any electric propulsion power source (nuclear, solar, or beamed power). Unique power management development needs for electric propulsion are (1) minimum mass and long life, (2) high power compared to space experience, i.e. megawatts instead of kilowatts, (3) fast arc suppression for protection of thrusters. Minimizing mass of power distribution leads to high distribution voltage and potential problems with plasma losses, arcing, and EMI. Thus while power management is a mature technology, the unique requirements of electric propulsion introduce a number of development risks beyond those usually experienced in space power systems.

Electric thruster technology has been under development since the beginning of the space program. Small thrusters are now operational, such as the resistance-heat-augmented hydrazine thrusters on certain communications spacecraft. Small arc and ion thrusters are nearing operational use for satellite stationkeeping.

Space transfer demands on electric propulsion performance place a premium on high power in the jet per unit mass of electric propulsion system. This in turn places a premium on thruster efficiency; power in the jet, not electrical power, propels spaceships. Space transfer electric propulsion also requires specific impulse in the range 5000 to 10,000 seconds. Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver this performance. Ion thrusters have acceptable efficiency but relatively low power per unit of ion beam emitting area. MPD thruster technology can deliver the needed Isp with high power per thruster, but has not yet reached efficiencies of interest. Circular ion thrusters have been built up to 50 cm diameter, with spherical segment ion beam grids. These can absorb on the order of 50 kWe each. A 10 MWe system would need 200 operating thrusters. The development alternatives all have significant risk: (1) Advance the state of the art of MPD thrusters to achieve high efficiency; (2) Develop propulsion systems with large numbers of thrusters and control systems; or (3) Advance the state of the art of ion thrusters to much larger size per thruster.

Nuclear power for electric propulsion - Space power reactor technology now under development (SP-100) may be adequate; needed advances are modest. Advanced power conversion systems are required to obtain power-to-mass ratios of interest. The SP-100 baseline is thermoelectric, which has no hope of meeting propulsion system performance needs. The most likely candidates are the closed Brayton (gas) cycle and the potassium Rankine (liquid/vapor) cycle. (Potassium provides the best match of liquid/vapor fluid properties to desired cycle temperatures.) Stirling cycle, thermionics, and a high-temperature thermally-driven fuel cell are possibilities. The basic technology for Brayton and Rankine cycles are mature; both are in widespread industrial use. Prototype space power Brayton and Rankine turbines have run successfully for thousands of hours in laboratories. The development risk here is that these are very complex systems; there is no experience base for coupling a space power reactor to a dynamic power conversion cycle;

there is no space power experience base at the power levels needed; and these systems, at power levels of interest for SEI space transfer application, are large enough to require in-space assembly and checkout. Space welding will be required for fluid systems assembly.

Solar power for space transfer propulsion - Solar power systems for space propulsion must attain much higher power-to-mass ratios than heretofore achieved. This implies a combination of advanced solar cells, probably multi-band-gap, and lightweight structural support systems. Required array areas are very large. Low-cost arrays, e.g. $100/watt, are necessary for affordable system costs, and automated construction of the large area structures, arrays, and power distribution systems appears also necessary. Where the nuclear electric systems are high development risk because of complexity and the lack of experience base at relevant power levels and with the space power conversion technologies, most of the solar power risk appears as technology advancement risk. If the technology advancements can be demonstrated, development risk appears moderate.

Avionics and software - Avionics and software requirements for space transfer systems are generally within the state of the art. New capability needs are mainly in the area of vehicle and subsystem health monitoring. This is in part an integration problem, but new techniques such as expert and neural systems are likely to play an important role.

An important factor in avionics and software development is that several vehicle elements having similar requirements will be developed, some concurrently. A major reduction in cost and integration risk for avionics can be achieved by advanced development of a "standard" avionics and software suite, from which all vehicle elements would depart.

Further significant cost savings are expected from advancements in software development methods and environments.

Environmental Control and Life Support (ECLS) - The main development risk in ECLS is for the Mars transfer habitat system. Other SEI space transfer systems have short enough operating durations that shuttle and Space Station Freedom ECLS system derivatives will be adequate. The Mars transfer requirement is for a highly closed physio-chemical system capable of 3 years' safe and dependable operation without resupply from Earth. The development risk arises from the necessity to demonstrate long life operation with high confidence; this may be expensive in cost and development schedule.

Man-Rating Approach

Man-rating includes three elements: (1) Design of systems to manned flight failure tolerance standards, (2) Qualification of subsystems according to normal man-rating requirements, and (3) Flight demonstration of critical performance capabilities and functions prior to placing crews at risk. Several briefing charts follow: the first summarizes a recommended approach and lists the subsystems and elements for which man-rating is needed; subsequent charts present recommended man-rating plans.

Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbital launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks.

Earth launch - The Earth launch risk to in-space transportation is the risk of losing a payload because of a launch failure. Assembly sequences are arranged to minimize the impact of a loss, and schedules include allowances for one make-up launch each mission opportunity.

Assembly and Orbital Launch Operations - Four sub-areas are covered: assembly, test and on-orbit checkout, debris, and inadvertent re-entry.

Assembly operations risk is reduced by verifying interfaces on the ground prior to launch of elements. Assembly operations equipment such as robot arms and manipulators will undergo space testing at the node to qualify critical capabilities and performance prior to initiating assembly operations on an actual vehicle.

Assembly risk varies widely with space transfer technology. Nuclear thermal rocket vehicles appear to pose minimum assembly risk; cryo/aerobraking are intermediate, and nuclear and solar electric systems pose the highest risk.

Test and on-orbit checkout must deal with consequences of test failures and equipment failures. This risk is difficult to quantify with the present state of knowledge. Indications are: (1) large space transfer systems will experience several failures or anomalies per day. Dealing with failures and anomalies must be a routine, not exceptional, part of the operations or the operations will not be able to launch space transfer systems from orbit; (2)

vehicles must have highly capable self-test systems and must be designed for repair, remove and replace by robotics where possible and for ease of repair by people where robotics cannot do the job; (3) test and on-orbit checkout will run concurrently with propellant loading and launch countdowns. These cannot take place on Space Station Freedom. Since the most difficult part of the assembly, test and checkout job must take place off Space Station Freedom the rest of the job probably should also.

Orbital debris presents risk to on-orbit operations. Probabilities of collision are large for SEI-class space transfer systems in low Earth orbit for typical durations of a year or more. Shielding is mandatory. The shielding should be designed to be removed before orbital launch and used again on the next assembly project.

Creation of debris must also be dealt with. This means that (1) debris shielding should be designed to minimize creation of additional debris, especially particles of dangerous size, and (2) operations need to be rigorously controlled to prevent an inadvertent loss of tools and equipment that will become a debris hazard.

Inadvertent re-entry is a low but possible risk. Some of the systems, especially electric propulsion systems, can have very low ballistic coefficient and therefore rapid orbital decay rate. Any of the SEI space transfer systems will have moderately low ballistic coefficient when not loaded with propellant. While design details are not far enough along to make a quantitative assessment, parts of these vehicles would probably survive reentry to become ground impact hazards in case of inadvertent reentry. For nuclear systems, it will be necessary to provide special support systems and infrastructure to drive the probability of inadvertent reentry to extremely low levels.

Launch Windows - Launch windows for single-burn high-thrust departures from low Earth orbit are no more than a few days because regression of the parking orbit line of nodes causes relatively rapid misalignment of the orbit plane and departure vector. For lunar missions, windows recur at about 9-day intervals.

For Mars, the recurrence is less frequent, and the interplanetary window only lasts 30 to 60 days. It is important to enable Mars launch from orbit during the entire interplanetary window. Three-impulse Mars departures make this possible; a plane change at apogee of the intermediate parking orbit provides alignment with the departure vector. Further analysis of the three-burn scheme is needed to assess penalties and identify circumstances where it does not work.

Launch window problems are generally minimal for low-thrust (electric propulsion) systems.

Mission Risk - Comparative mission risk was analyzed by building risk trees and performing semi-quantitative analysis. The next chart presents a comparison of several mission modes; after that are the risk trees for these modes.

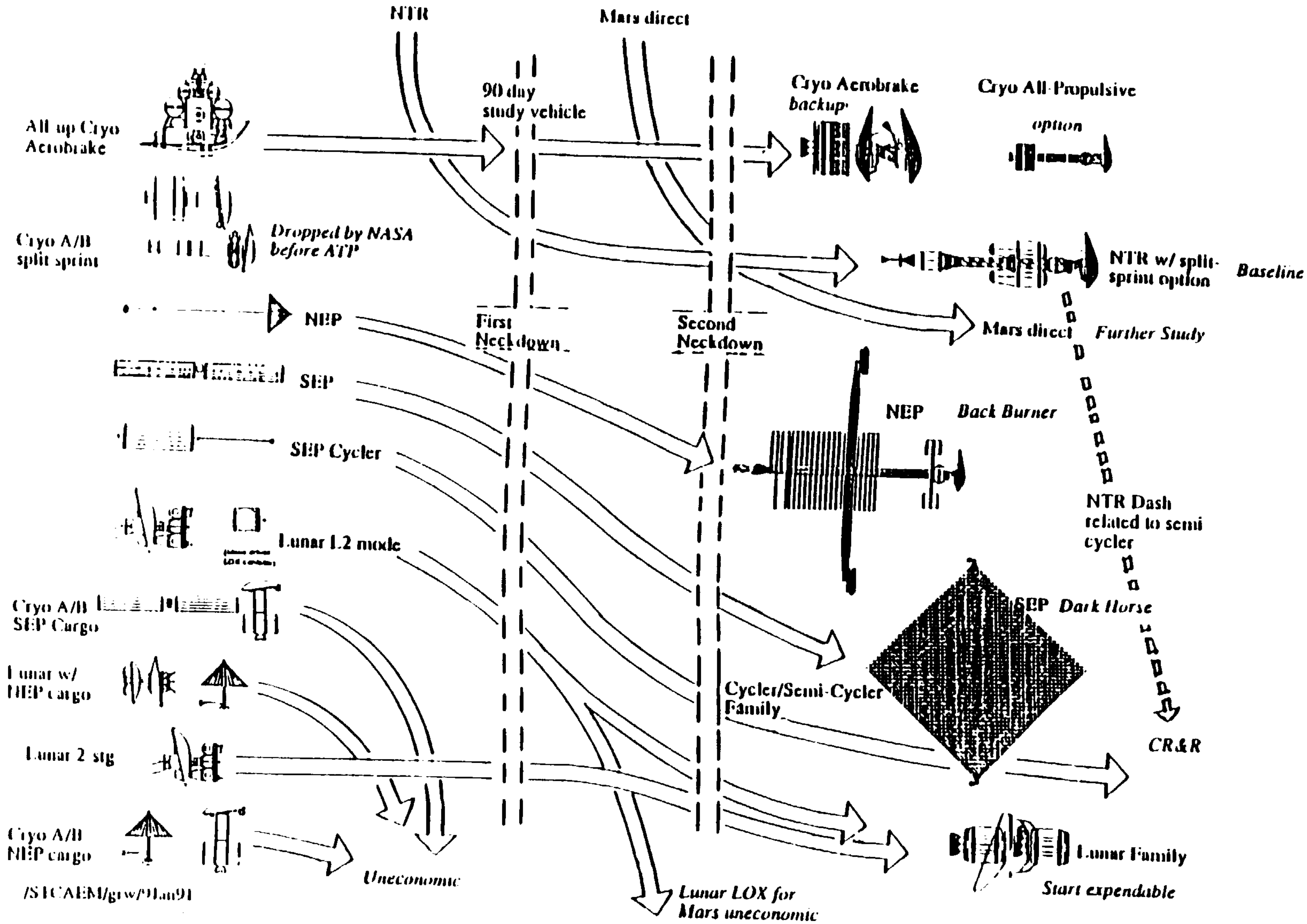
Ionizing Radiations and Zero G - The threat from ionizing radiations is presented elsewhere in this document. Presented here are the mitigating strategies for ionizing radiations and zero g.

Nuclear systems operations present little risk to flight crews. Studies by University of Texas at Austin showed that radiation dose to a space station crew from departing nuclear vehicles is very small provided that sensible launch and flight strategies are used. On-board crews are protected by suitable shielding and by arrangement of the vehicle, i.e. hardware and propellant between reactors and the crew and adequate separation distances. After nuclear engines are shut off, radiation levels drop rapidly so that maneuvers such as departure or return of a Mars excursion vehicle are not a problem. On-orbit operations around a returned nuclear vehicle are deferred until a month or two after shutdown, by which time radioactivity of the engine is greatly reduced.

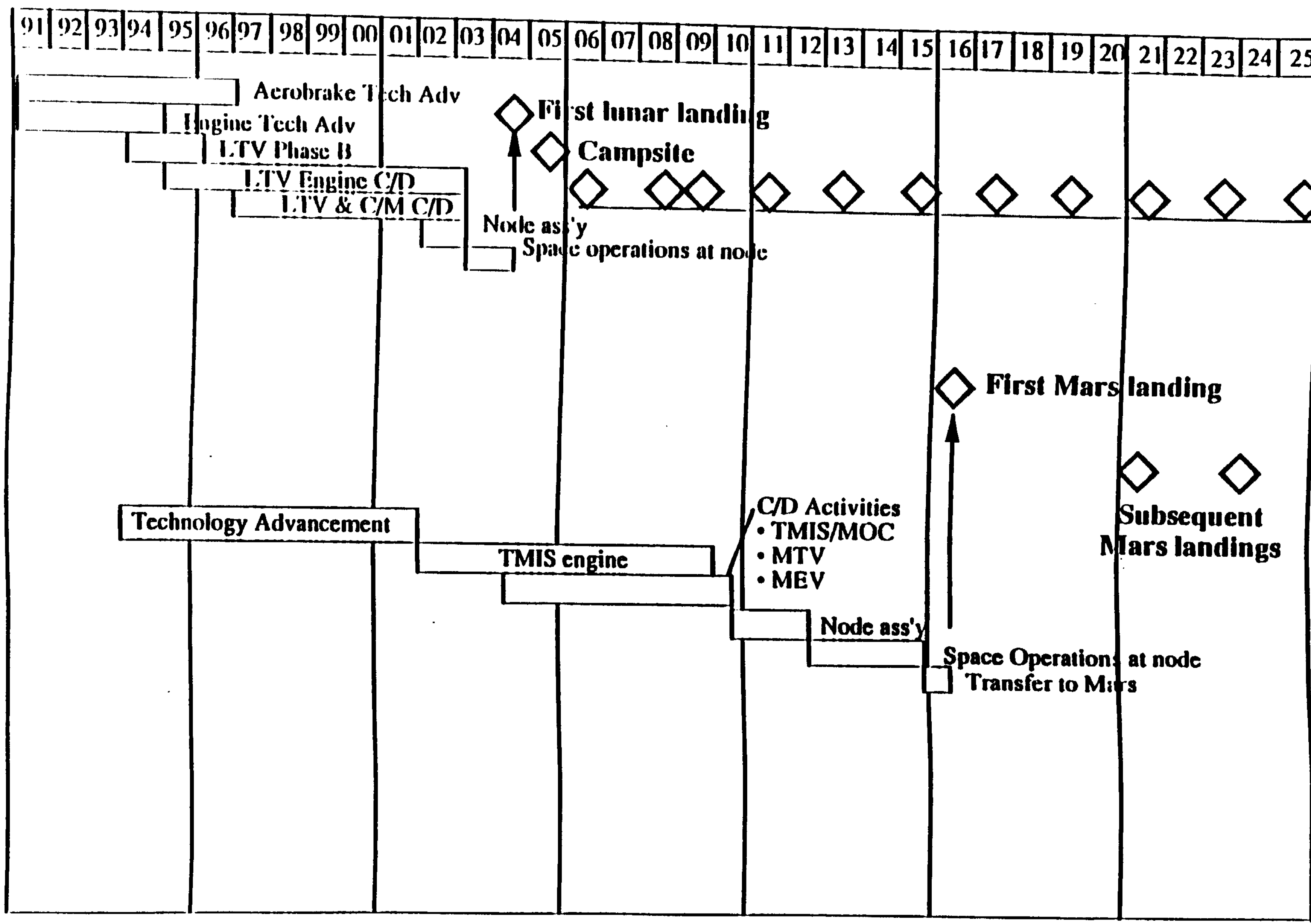
Reactor disposal has not been completely studied. Options include solar system escape and parking in stable heliocentric orbits between Earth and Venus.

Crew radiation dose abatement employs "storm shelters" for solar flares, and either added shielding of the entire vehicle or fast transfers (or both) to reduce galactic cosmic ray exposure. Assessments are in progress; tradeoffs of shielding versus fast trips have yet to be completed. Expected impact for lunar missions is negligible and for Mars missions, modest.

Overall Study Flow



Minimum Program



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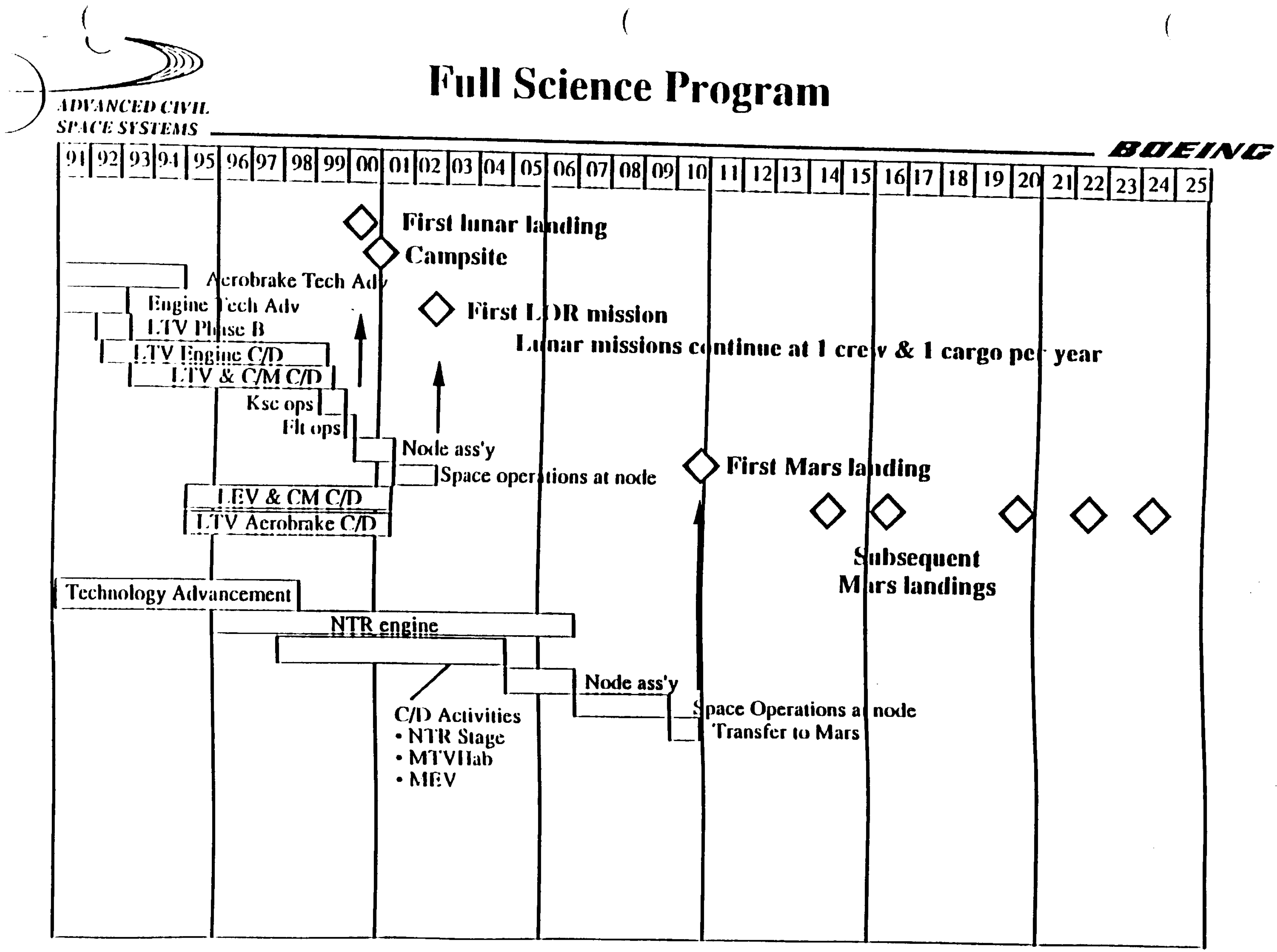

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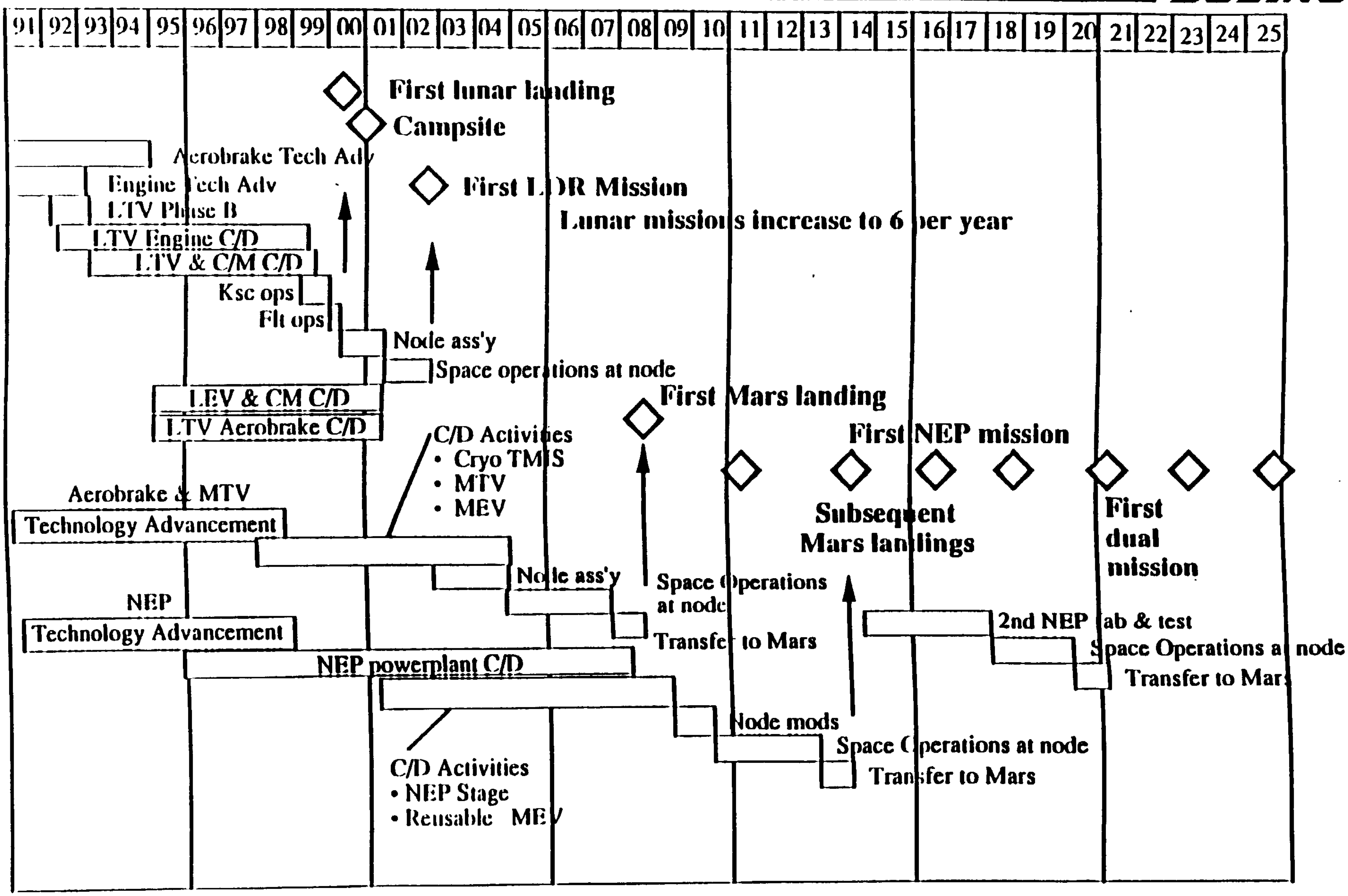
ADVANCED CIVIL SPACE SYSTEMS

Industrialization and Settlement Program

BOEING



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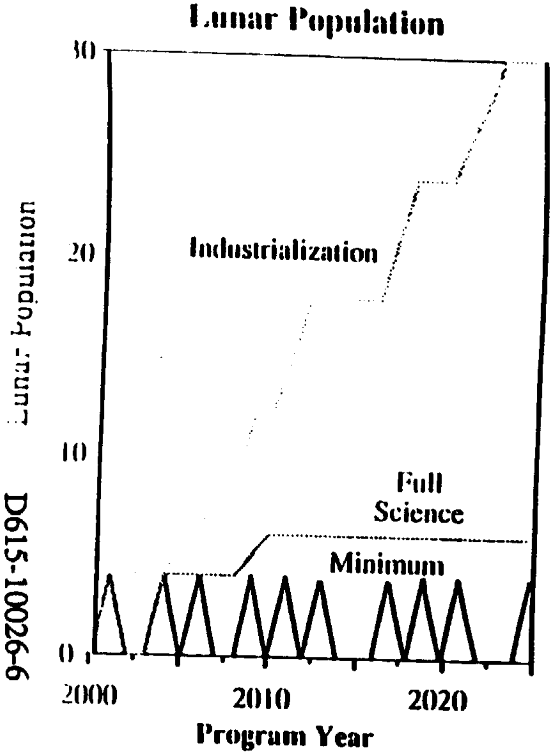
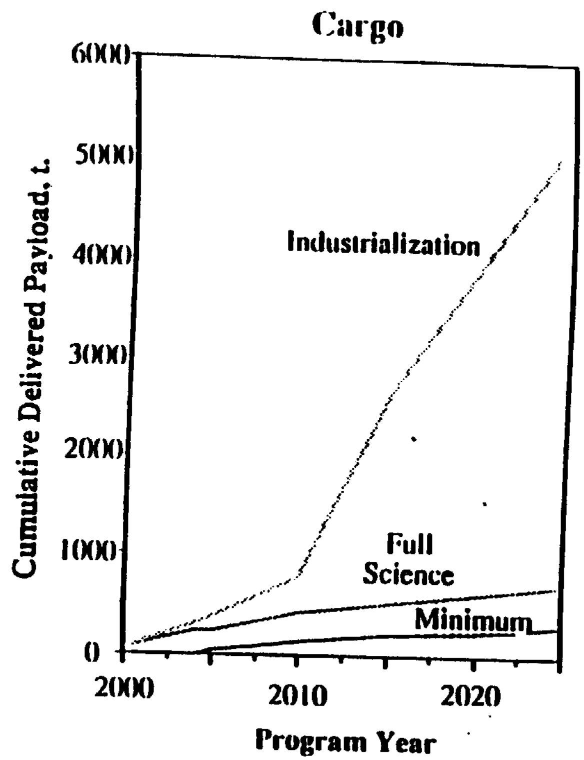
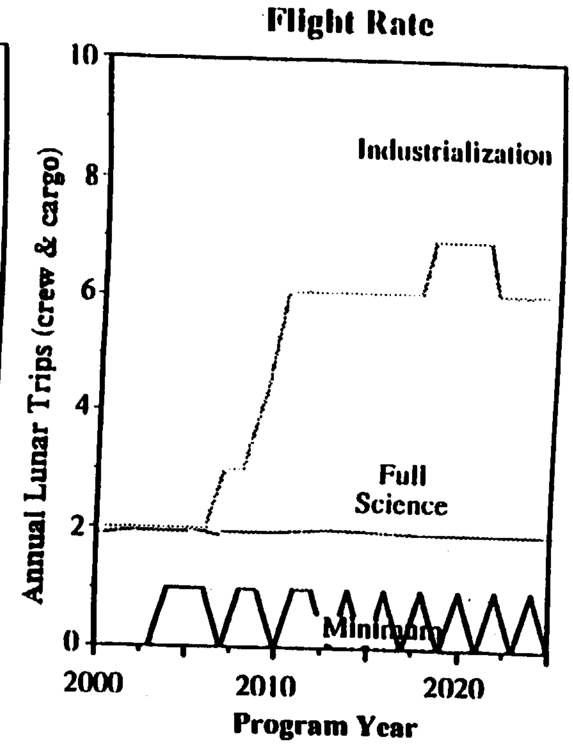
Lunar Program Comparison







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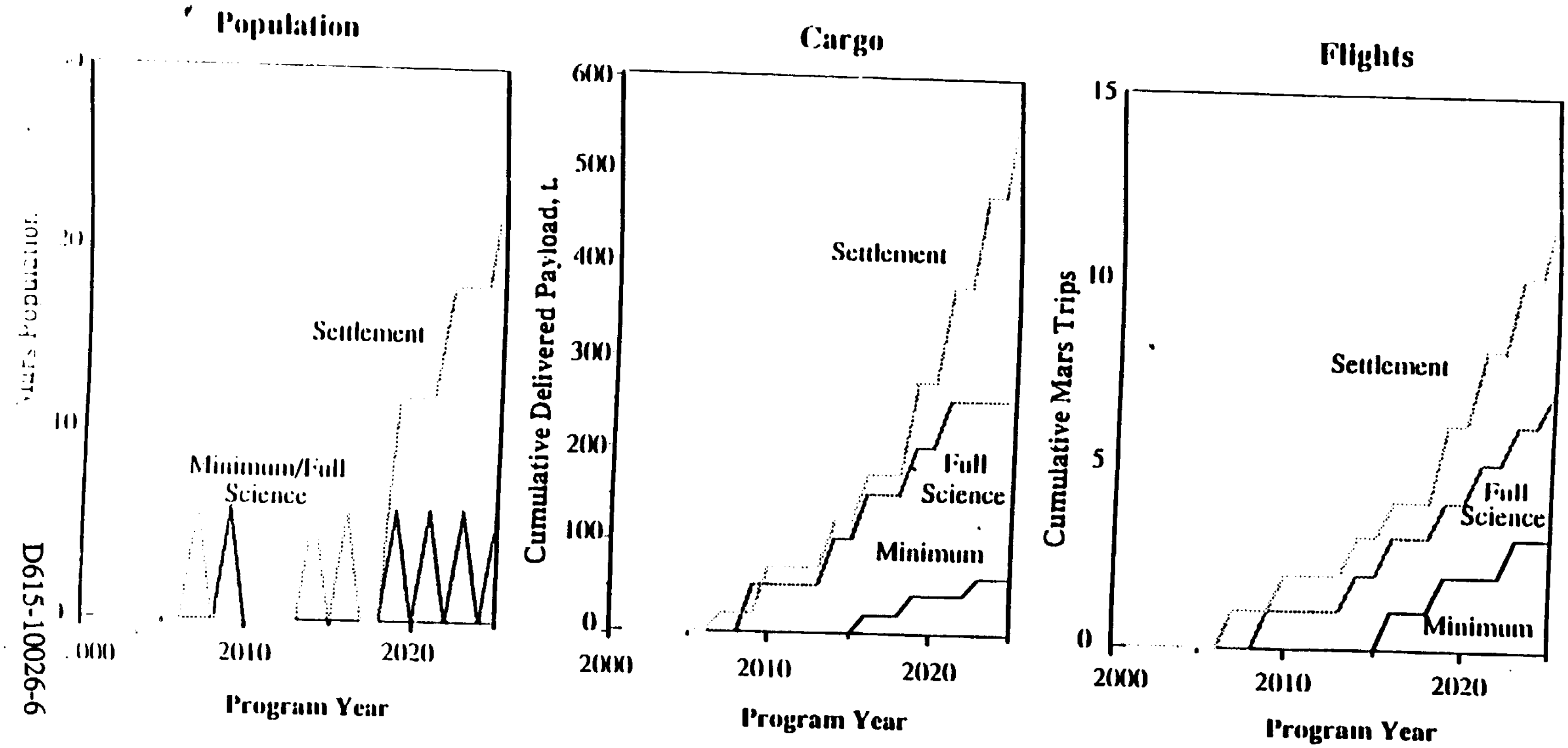
Program Comparisons



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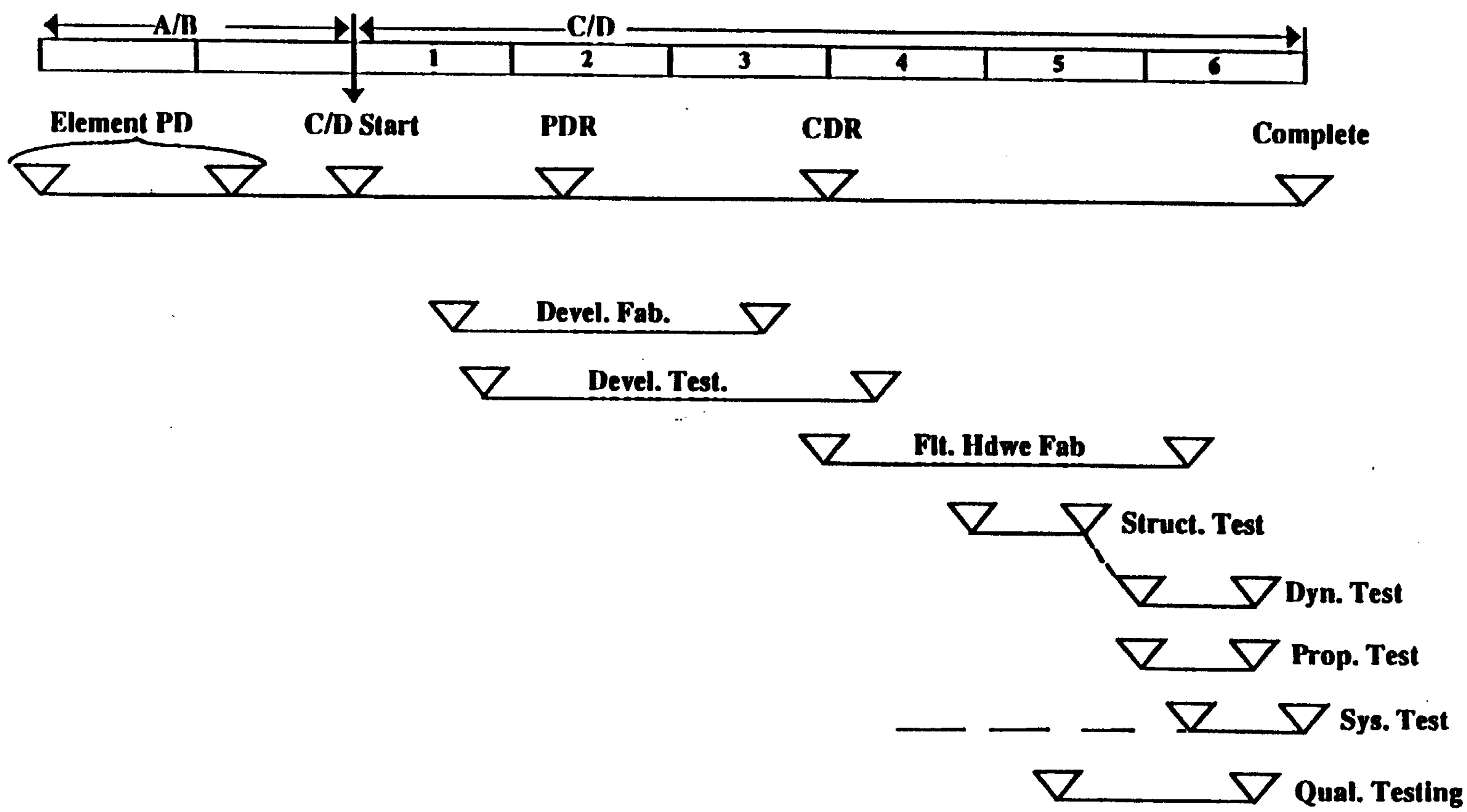
Reference 6 yr Full-Scale Development Schedule



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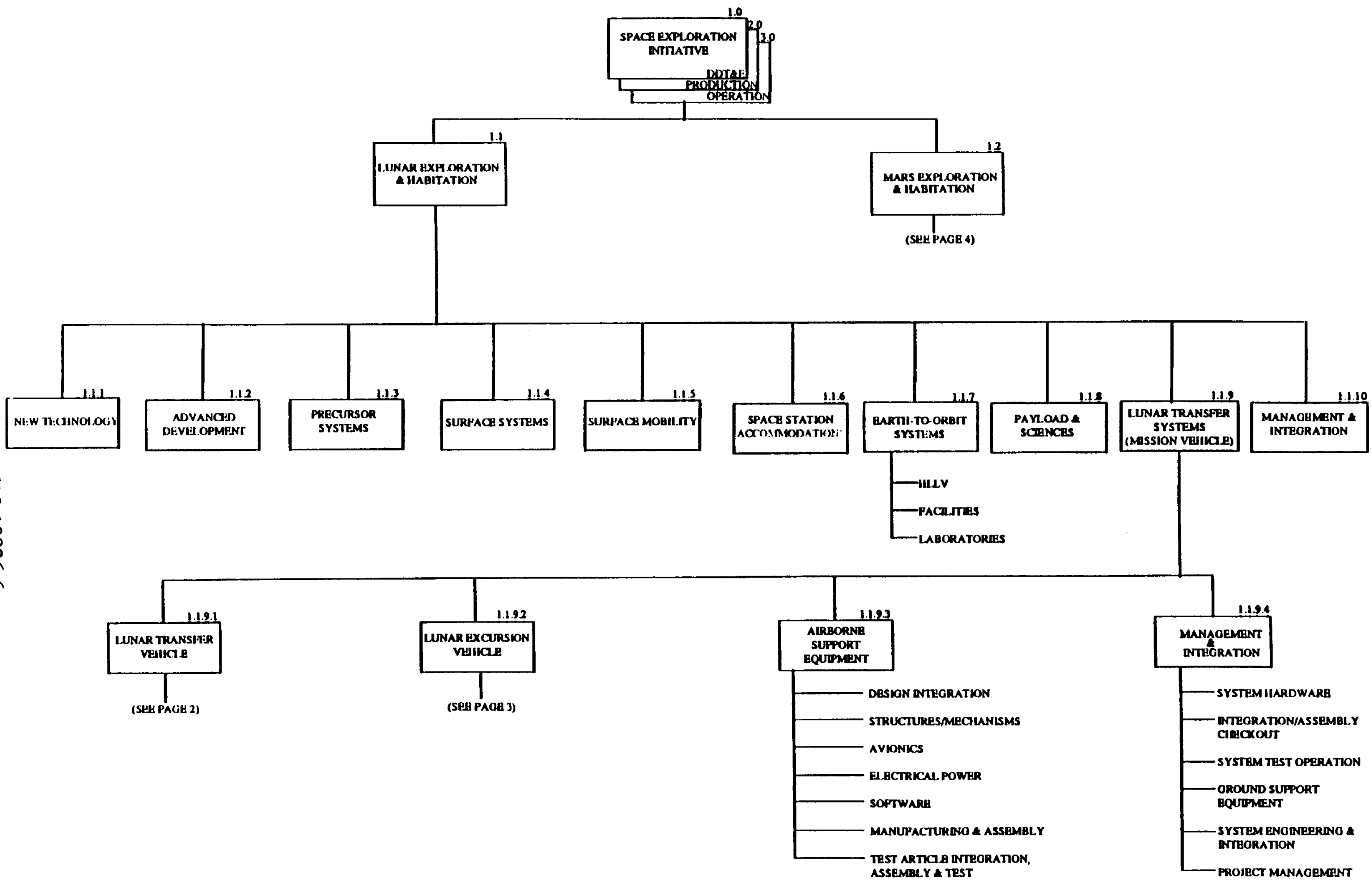
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS

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SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS

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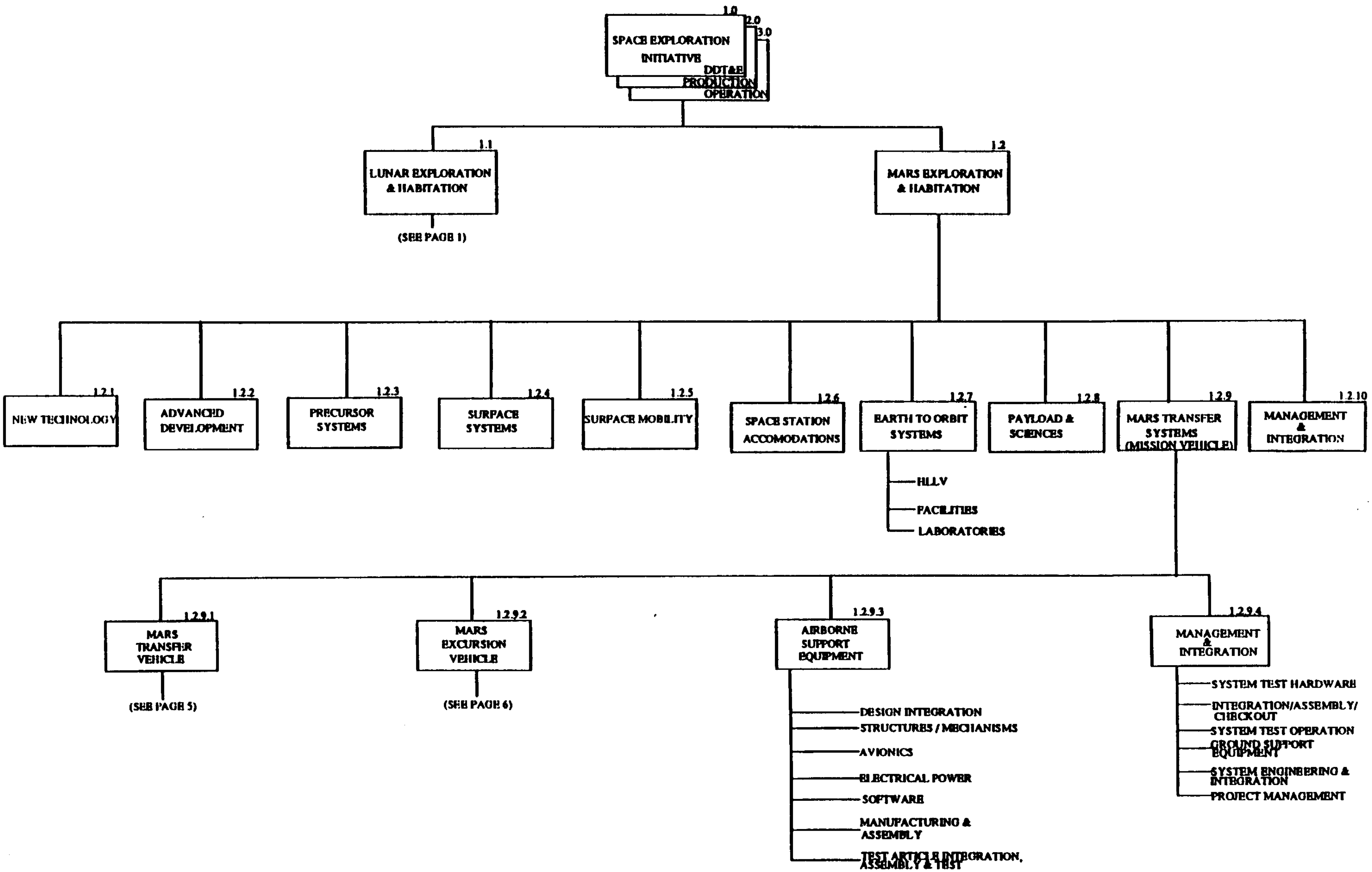
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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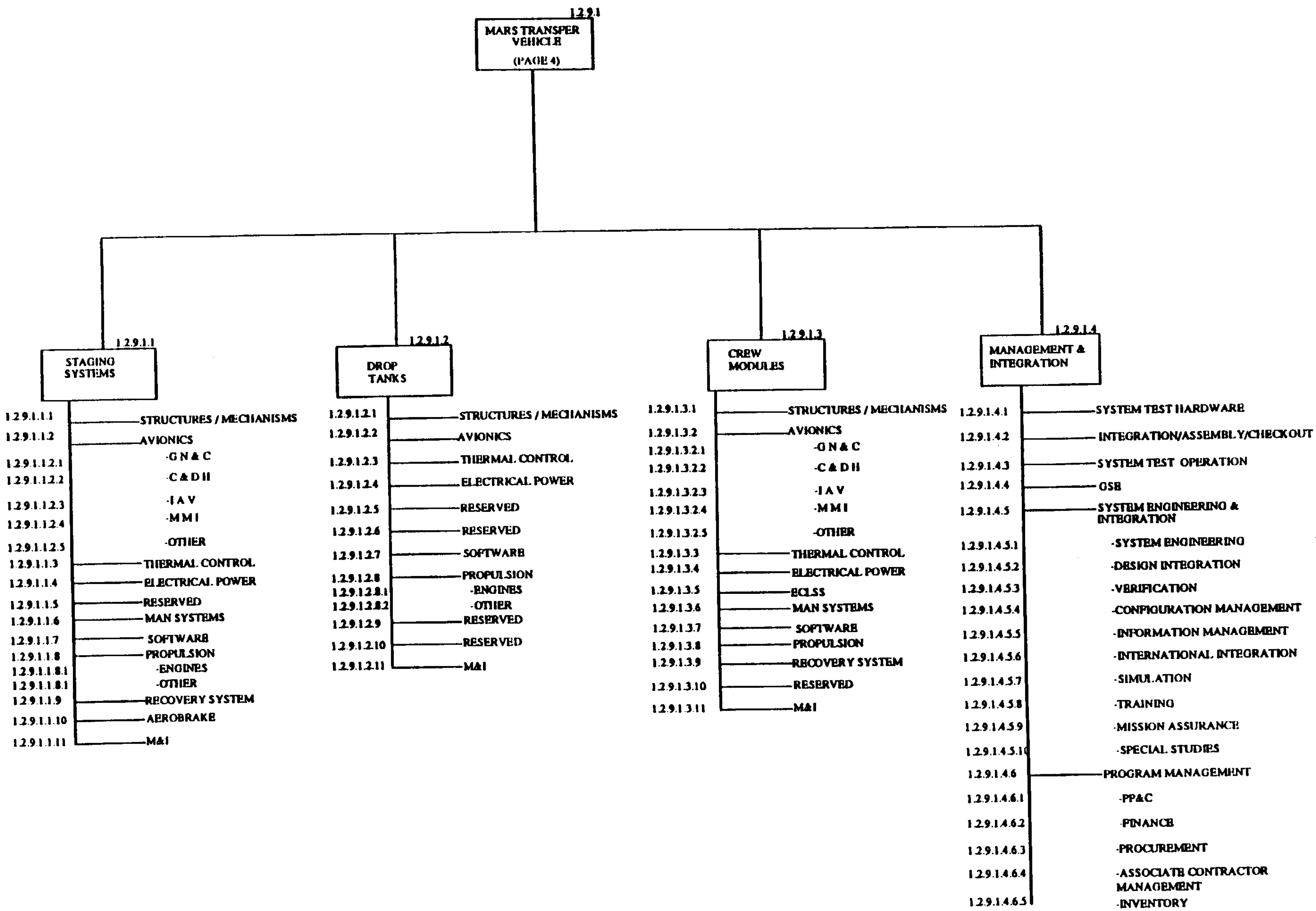
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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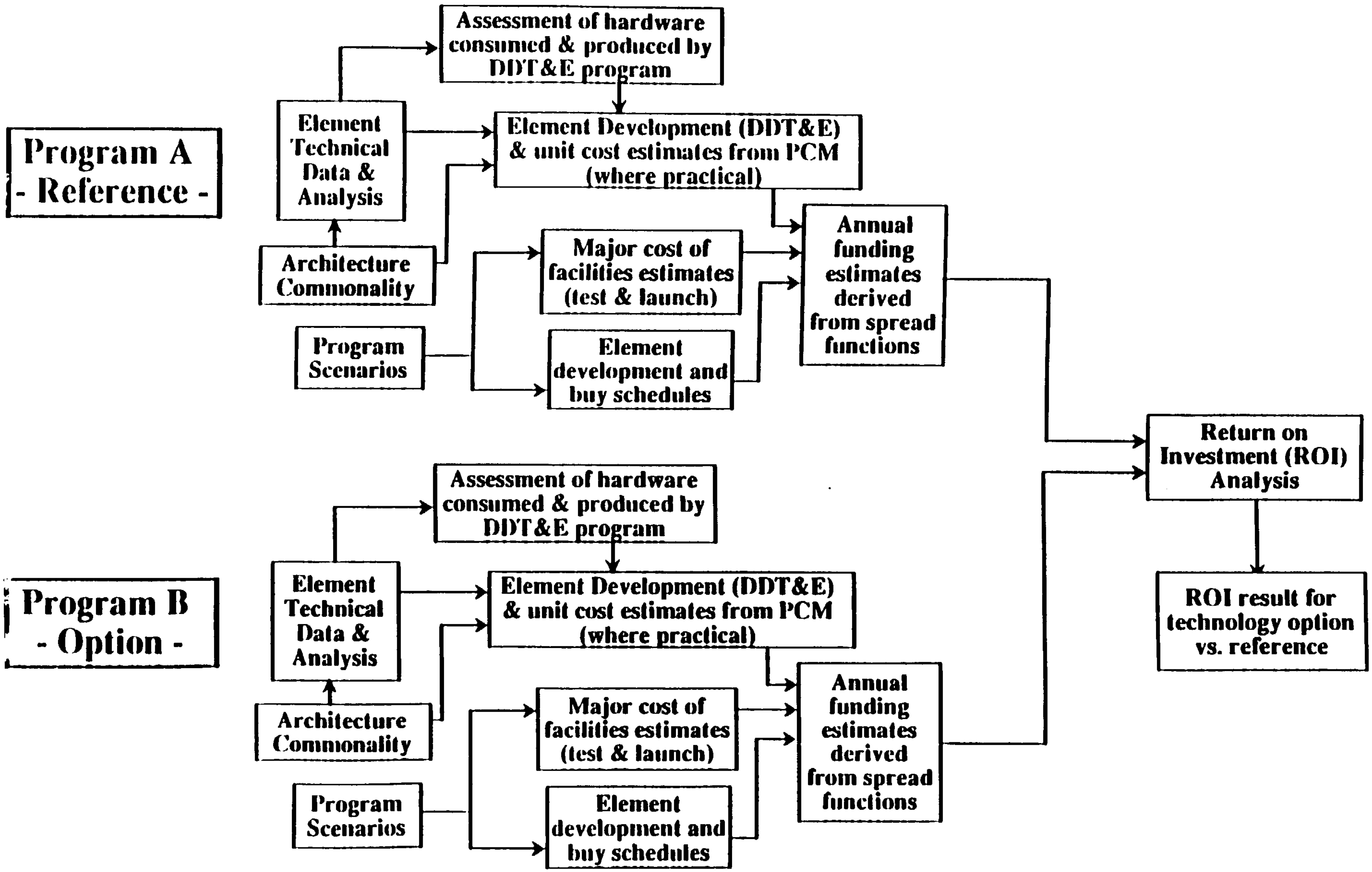
Costing Methodology Flow



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Boeing Parametric Cost Model (PCM)

Features

- Designed specifically for advanced system estimating
- Uses company-wide, uniform computerized data base
- Contains historical data compiled since 1969
- Allows direct input of known costs into the estimate

Main Inputs	Results
• Hardware Characteristics - Category (e.g., primary structure, power conditioning, etc.) - Weight (or Thrust) - Complexity - % Off-the-Shelf - Maturity - Quantity - Manufacturing Learning Curve • Support Cost Factors - Systems Engineering - Management - Operations - Spares	• DDT&E and Manufacturing Estimates - Based on previous Boeing programs - Provides first flight unit costs - Excludes test hardware - Excludes fees • New hardware must be relatable to PCM database to produce reasonable estimate • PCM estimates improve with increasing hardware detail.

Components		*Lunar/Mars*		
		Minimum	Full Science	Settle /Ind
HLLV	Cargo Carrier & Core	X	X	X
	STME	X	X	X
	Recov PA Mod	X	X	X
Propulsion	Std Avionics Suite	X	X	X
	Adv Space Engine	X	X	X
	NTR Tanks		X	
	MOC Tank	X		X
	MOC Core	X		X
	NTR Stage		X	
	NTR Engine		X	
	NEP Stage			X
	NEP Engine			X
	TMIS Engine	X		X
	TMIS Tank	X		X
	TMIS Core	X		X
Modules	LEO Tanker	X	X	X
	LTV Hab	X	X	X
	LTV	X	X	X
	LEV	X	X	X
	LEV Crew Module	X	X	X
	MTV	X		X
	MTV Crew Module	X	X	X
	MEV	X	X	X
	RMEV			X
	mini-MEV		X	
	MEV Crew Module	X	X	X
	Lunar Aerobrake	X		
	MTV Aerobrake			
	MEV Aeroshell	X	X	X
	MCRV	X	X	X

ADVANCED CIVIL SPACE SYSTEMS

LTV Preliminary PCM Summary

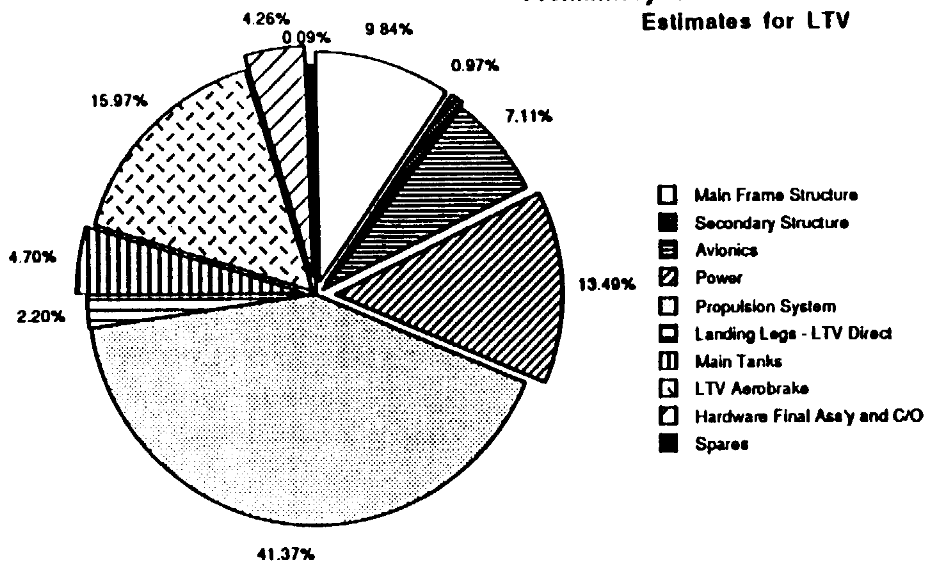
BOEING

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Main Frame Structure	52.714	23.327	76.040
Secondary Structure	4.352	3.150	7.502
Avionics	29.207	25.737	54.944
Power	79.540	24.755	104.295
Propulsion System - LTV	247.994	71.867	319.861
Landing Legs - LTV Direct	11.494	5.504	16.998
Main Tanks	8.927	27.426	36.352
LTV Aerobrake	85.535	37.944	123.479
Hardware Final Ass'y and C/O	----------	32.956	32.956
Spares	----------	0.659	0.659
Hardware Total Costs	519.762	253.325	773.087
System Engineering & Integration	84.004	----------	84.004
Software Engineering	62.909	----------	62.909
Systems Ground Test Conduct	183.580	----------	183.580
Systems Flight Test Conduct	----------	----------	----------
Peculiar Support Equipment	102.932	12.633	115.565
Tooling & Special Test Equipment	----------	54.152	54.152
Task Direct Quality Assurance	----------	24.379	24.379
Logistics	21.892	----------	21.892
Liaison Engineering	25.281	----------	25.281
Data	9.093	----------	9.093
Training	O/H	----------	----------
Facilities Engineering	O/H	----------	----------
Safety	O/H	----------	----------
Graphics	O/H	----------	----------
Outplant	O/H	----------	----------
Program Management	O/H	----------	----------
Support Effort Total	489.690	91.164	580.854
Total Estimate	**1009.452**	**344.489**	**1353.941**

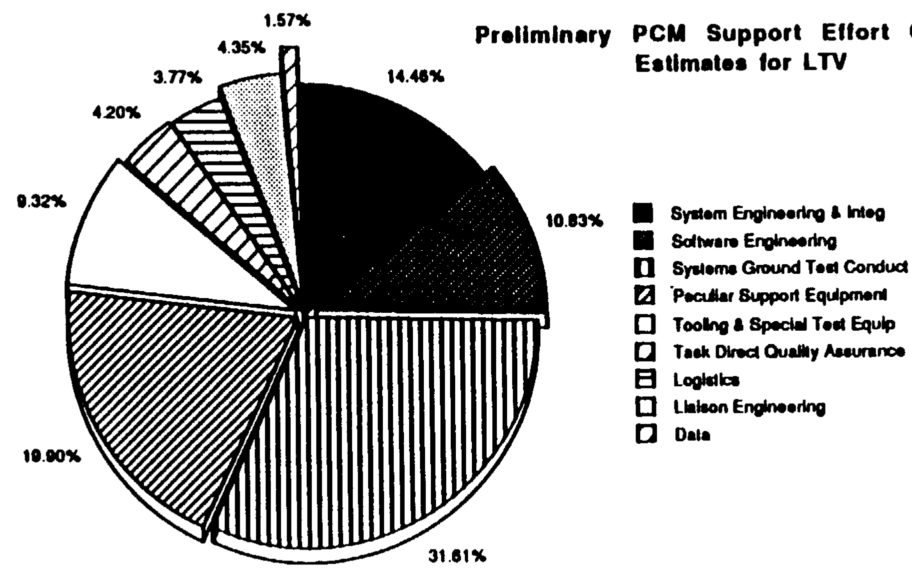
O/H = Overhead charge (included in above costs)

LTV Preliminary PCM Summary - continued

Preliminary PCM Hardware Cost Estimates for LTV



Preliminary PCM Support Effort Cost Estimates for LTV



ADVANCED CIVIL SPACE SYSTEMS

LTV Crew Module Preliminary PCM Summary

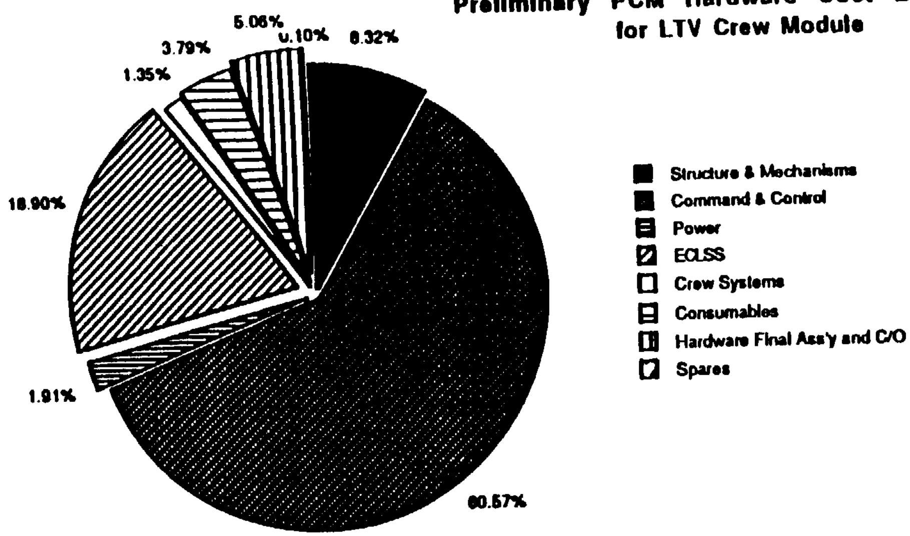
BOEING

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Structure and Mechanisms	49.421	32.340	81.760
Command & Control	391.648	203.804	595.452
Power	10.827	7.932	18.760
ECLSS	121.176	64.672	185.849
Crew Systems	7.481	5.815	13.297
Consumables	20.076	17.178	37.253
Hardware Final Ass'y and C/O	----------	49.761	49.761
Spares	----------	0.995	0.995
Hardware Total Costs	600.629	382.498	983.126
System Engineering & Integration	90.896	----------	90.896
Software Engineering	67.794	----------	67.794
Systems Ground Test Conduct	204.168	----------	204.168
Systems Flight Test Conduct	----------	----------	----------
Peculiar Support Equipment	116.953	19.075	136.028
Tooling & Special Test Equipment	----------	87.085	87.085
Task Direct Quality Assurance	----------	36.811	36.811
Logistics	23.841	----------	23.841
Liaison Engineering	30.280	----------	30.280
Data	10.302	----------	10.302
Training	O/H	----------	----------
Facilities Engineering	O/H	----------	----------
Safety	O/H	----------	----------
Graphics	O/H	----------	----------
Outplant	O/H	----------	----------
Program Management	O/H	----------	----------
Support Effort Total	544.233	142.971	687.204
Total Estimate	**1144.862**	**525.469**	**1670.331**

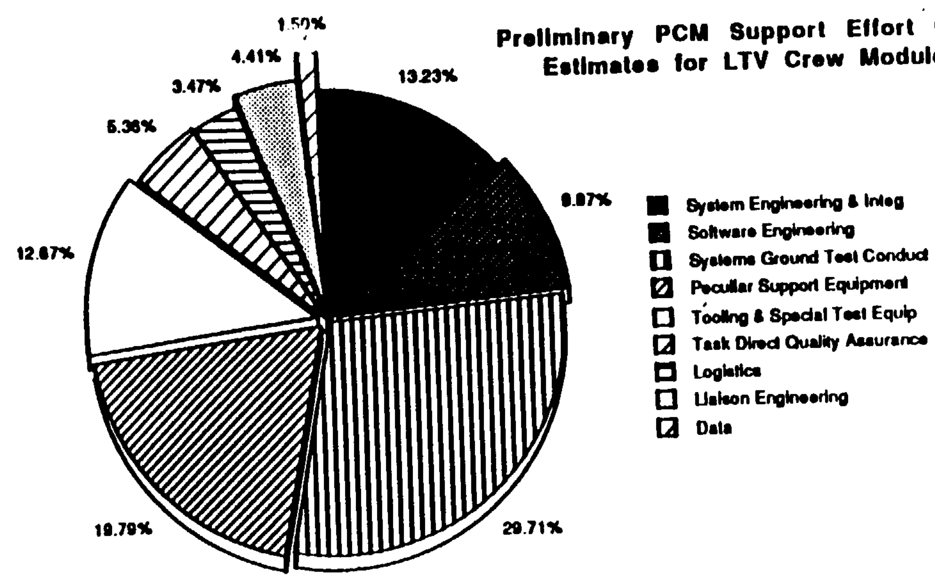
O/H = Overhead charge (included in above costs)

LTV Crew Module Preliminary PCM Summary - continued

Preliminary PCM Hardware Cost Estimat for LTV Crew Module



Preliminary PCM Support Effort Cost Estimates for LTV Crew Module



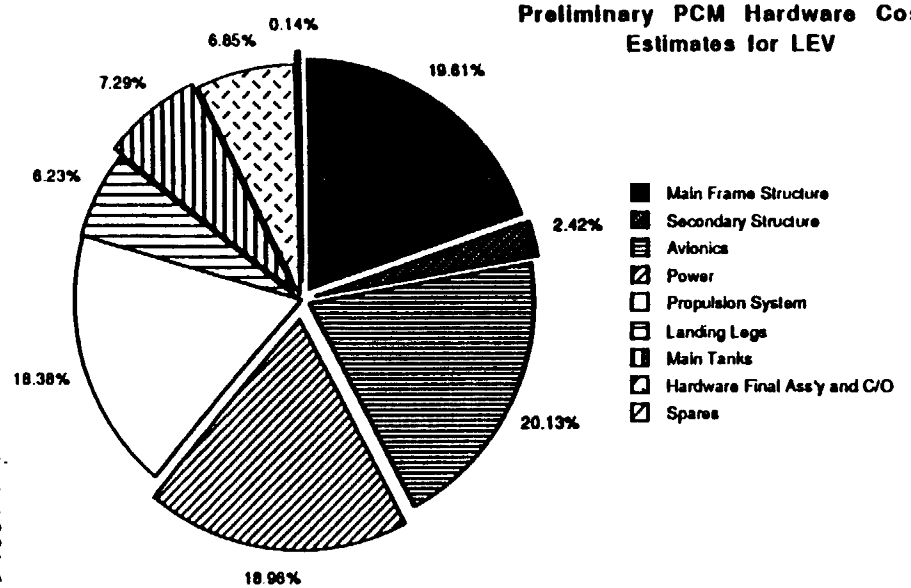
LEV Preliminary PCM Summary

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Main Frame Structure	38.435	15.098	53.533
Secondary Structure	3.934	2.684	6.618
Avionics	29.207	25.737	54.944
Power	39.733	12.022	51.755
Propulsion System - LEV	----------	50.167	50.167
Landing Legs	11.494	5.504	16.998
Main Tanks	6.541	13.372	19.913
Hardware Final Ass'y and C/O	----------	18.688	18.688
Spares	----------	0.374	0.374
Hardware Total Costs	129.344	143.646	272.990
System Engineering & Integration	28.807	----------	28.807
Software Engineering	23.000	----------	23.000
Systems Ground Test Conduct	43.383	----------	43.383
Systems Flight Test Conduct	----------	----------	----------
Peculiar Support Equipment	23.784	7.164	30.948
Tooling & Special Test Equipment	----------	28.154	28.154
Task Direct Quality Assurance	----------	13.824	13.824
Logistics	6.876	----------	6.876
Liaison Engineering	4.266	----------	4.266
Data	2.404	----------	2.404
Training	O/H	----------	----------
Facilities Engineering	O/H	----------	----------
Safety	O/H	----------	----------
Graphics	O/H	----------	----------
Outplant	O/H	----------	----------
Program Management	O/H	----------	----------
Support Effort Total	132.521	49.142	181.663
Total Estimate	**261.865**	**192.788**	**454.652**

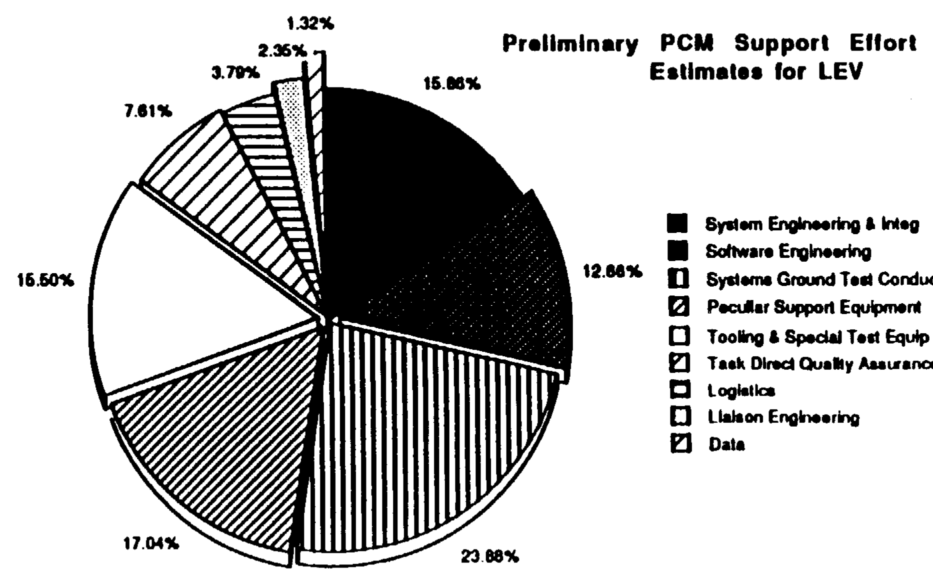
O/H = Overhead charge (included in above costs)

LEV Preliminary PCM Summary - continued

Preliminary PCM Hardware Cost Estimates for LEV



Preliminary PCM Support Effort Cost Estimates for LEV



ADVANCED CIVIL SPACE SYSTEMS

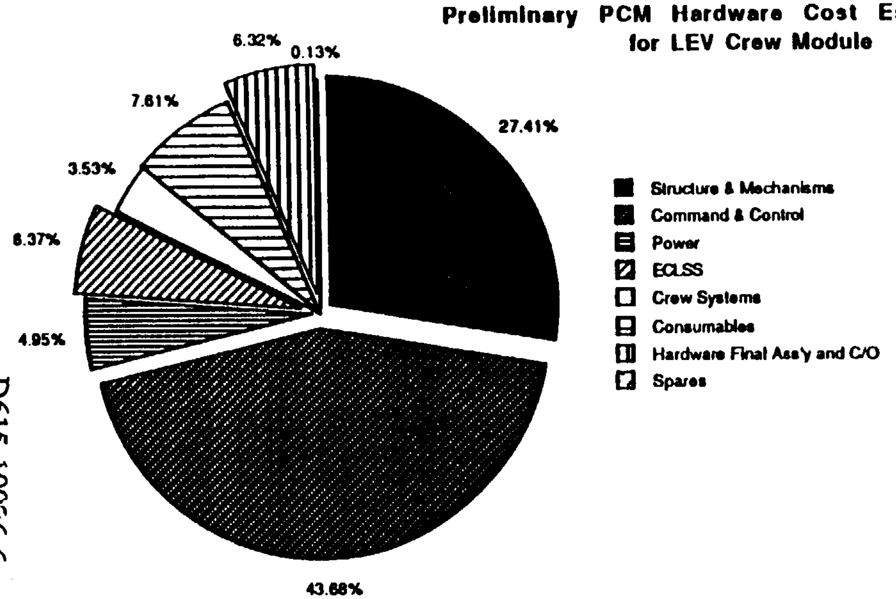
LEV Crew Module Preliminary PCM Summary

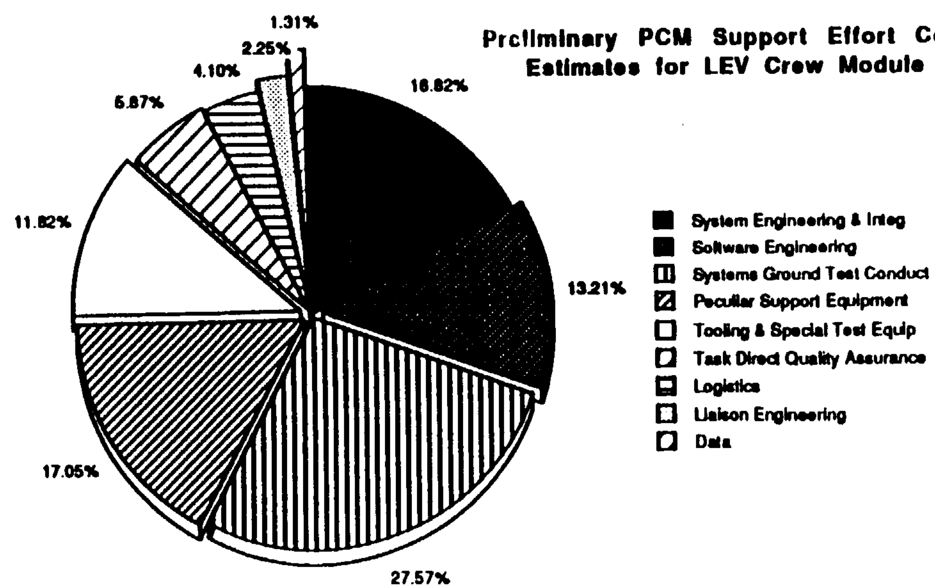
BOEING

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Structure and Mechanisms	46.200	29.088	75.288
Command & Control	60.611	59.348	119.959
Power	5.960	7.623	13.583
ECLSS	10.449	7.049	17.498
Crew Systems	5.754	3.948	9.702
Consumables	12.193	8.696	20.889
Hardware Final Ass'y and C/O	----------	17.363	17.363
Spares	----------	0.347	0.347
Hardware Total Costs	141.167	133.462	274.628
System Engineering & Integration	36.785	----------	36.785
Software Engineering	28.905	----------	28.905
Systems Ground Test Conduct	60.317	----------	60.317
Systems Flight Test Conduct	----------	----------	----------
Peculiar Support Equipment	30.653	6.656	37.309
Tooling & Special Test Equipment	----------	25.865	25.865
Task Direct Quality Assurance	----------	12.844	12.844
Logistics	8.959	----------	8.959
Liaison Engineering	4.912	----------	4.912
Data	2.862	----------	2.862
Training	O/H	----------	----------
Facilities Engineering	O/H	----------	----------
Safety	O/H	----------	----------
Graphics	O/H	----------	----------
Outplant	O/H	----------	----------
Program Management	O/H	----------	----------
Support Effort Total	173.392	45.365	218.757
Total Estimate	**314.559**	**178.827**	**493.385**

O/H = Overhead charge (included in above costs)

LEV Crew Module Preliminary PCM Summary - continued





	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	LECM	141	0	2.413	340.233	153	3.5	
3	LTCM	601	0	2.05	1232.05	438	3.65	
4	LTV	358	0	2.05	733.9	252	3.5	
5	LTV engine	161	0	2.05	330.05	8	24	
6	LEV	129	70	2.24	445.76	143	3.5	
7	LTV w/o A/B	273	0	2.05	559.65	214	3.5	
8	EOCV	280		2.7	756	214	3.65	
9					0			
10					0			

Lunar Cost Buildup

	I	J	K	L	M	N	O
1	DDT&E no Fe	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/	Fee
2	875.733	8	945.79164	1	153	165.24	
3	2830.75	8	3057.21	1	438	473.04	
4	1615.9	8	1745.172	1	252	272.16	
5	522.05	8	563.814	7	56	60.48	
6	946.26	8	1021.9608	1	143	154.44	
7	1308.65	8	1413.342	1	214	231.12	
8	1537.1	8	1660.068	1	214	231.12	
9	0		0		0	0	
10			10,407.3				1587.6

Page 2

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Development Risk Assessment For Aerobraking By Function

MISSION FUNCTION	BRAKE SIZE	ATMOSPHERE KNOWLEDGE & UNCERTAINTY	TARGET FOR ENTRY: GN&C PRECISION	HEATING/TPS	AERO PASS GN&C PRECISION REQUIRED
Lunar return Earth landing	Small, no ass'y required	Accurate knowledge, low uncert. effect	Very high	State-of-the-Art	State-of-the-Art
Lunar return Earth landing	Moderate requires assembly	Accurate knowledge, high uncert. effect	Very high	State-of-the-Art	Believed State-of-the-Art
Mars landing from orbit	Large, requires assembly	Poor knowledge, low uncert. effect	Can be high, e.g. done from Mars orbit	State-of-the-Art	Believed State-of-the-Art
Mars return Earth landing	Small, no ass'y required	Accurate knowledge, moderate uncertainty effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Accurate knowledge, high uncert. effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Poor knowledge, high uncert. effect	Poor, unless nav-aids in Mars orbit	High heating rates, some TPS advancement needed	Advancements required

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V. Commonality/Evolution

A. System Commonality

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V. Commonality/Evolution

A. Commonality - Having identified commonality as having high cost-effectiveness leverage, STCAEM developed an evolutionary strategy for facilitating it and benefitting from it. The greatest commonality challenge is between early systems designed for lunar missions and later systems designed for Mars missions. Because of similar flight regimes, crew systems requirements and durations, excursion vehicle crew cabins for lunar and Mars uses proved most amenable to strict commonality; for conceptual purposes the LEV and MEV crew cabs can be considered just slightly different "models" of the same element.

An effort was made to extend the commonality approach to entire excursion vehicle designs for the Moon and Mars. This extreme degree of commonality was found to appear conceptually feasible only for a particular size class of vehicle (a 25 t propellant-capacity vehicle typical of LEV designs is comparable to a so-called "mini-MEV", which would take 3 crew and 1 t payload to the surface of Mars). Except for this special case, the differing gravity levels of the Moon and Mars, and configuration complications arising from aerobraking upon descent at Mars, tend to drive LEV and MEV designs apart. The LEV and MEV are likely to come on line years apart in any case, and we found a more productive way to introduce commonality.

We found it useful to consider commonality at three program levels: (1) *mission design*, using the same mission design to accomplish different programmatic architectures (a problematic category because mission designs are by definition tied to unique requirements); (2) *functional element*, using end-items from the same production line to fill different roles within a given mission design (the most appropriate example we developed is the evolutionary LTF described earlier); and (3) *performing subsystem*, using system assemblies or components from the same production line in different functional elements (a sensible way to standardize industry, get predictable performance and facilitate product longevity). This latter approach, applied to engines, sensors, processors, some structural components and modular life support hardware, shows great promise for cost-effectiveness and preserving program resiliency. At the component level, extensive commonality can be worked into the fabric of SEI.

Another application of this subsystem/component commonality-for-evolution approach is the potential use of hardware systems developed mostly for lunar transportation, augmented by long-duration crew systems, for early Mars missions staged out of high-orbit node locations like Earth-Moon L2. TMI ΔV for this mission mode is such that the need for a large TMIS is obviated altogether, as is the need for a large cryogenic space engine. The use of chemically-propelled

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Key Architecture Drivers

Shown are two categories of architecture drivers which STCAEM has identified to have high leverage for both program lifecycle-cost reduction and for safety. Both drivers have been broken down into component principles, defined succinctly for STCAEM's purpose as shown. It is proposed that these key principles are valuable as both design guidelines and performance metrics, not only for successful exploration mission architectures and their components (elements and subsystems), but also for successful overall program strategies.

Key Architecture Drivers

Flexibility

Robustness is the ability to perform nominally despite variable or unanticipated conditions

Resiliency is the ability to recover from accidental delays or mishaps

Evolution is an adaptation over time to changing requirements

Multi-use Design

Re-usability means using the same hardware item more than once

Commonality means using the same hardware item in more than one setting

Commonality

The term "commonality" is common currency in the SEI community. It is vital to explore just what it means, what benefits it can introduce, what costs it can incur, and then to decide where and when it is feasible and desirable to strive for it. Shown is a suggested outline for levels of commonality.

Commonality of mission architectures does not make much sense. While very similar functional element designs can be used in distinct candidate mission architectures (i.e. the same Mars lander may be appropriate for either small or large scale architectures), a mission architecture by definition is unique to a particular combination of mission requirements. (Commonality among concept studies at the architecture level is of course desirable, but has little to do with ultimate hardware cost.)

Commonality at the subsystem level is a familiar approach, and if implemented at the prototyping stage can greatly facilitate cost-effective, robust systems design.

The problem is commonality at the functional element level. Simply stated, insisting on vehicle commonality for environments as different as introduced by lunar and Mars missions can easily lead to an end-item design whose performance in both environments is substantially compromised. A helpful refinement of the concept of functional element commonality is *options* assembled from a standardized kit of systems and subsystems. End items are then familial rather than identical, and their performance can be more flexible.

Commonality

"Mission design" level: **Use the same mission design to accomplish different programmatic architectures** (e.g. L2-based NEP with low-L/D MEVs for both max-science *and* Mars settlement programs)

- **Problematic** (scope, scale, timing and technology are closely goal-driven; detailed mission designs tend to be unique)

"Functional element" level : **Use end-items *from the same production line* to fill different roles within a given mission design** (e.g. common lunar/Mars excursion vehicle)

- **Common end-items need not be identical** (cars with different options are typically produced on the same line)
- **Procurement from the same vendor** (duplicating production lines defeats cost-reduction)

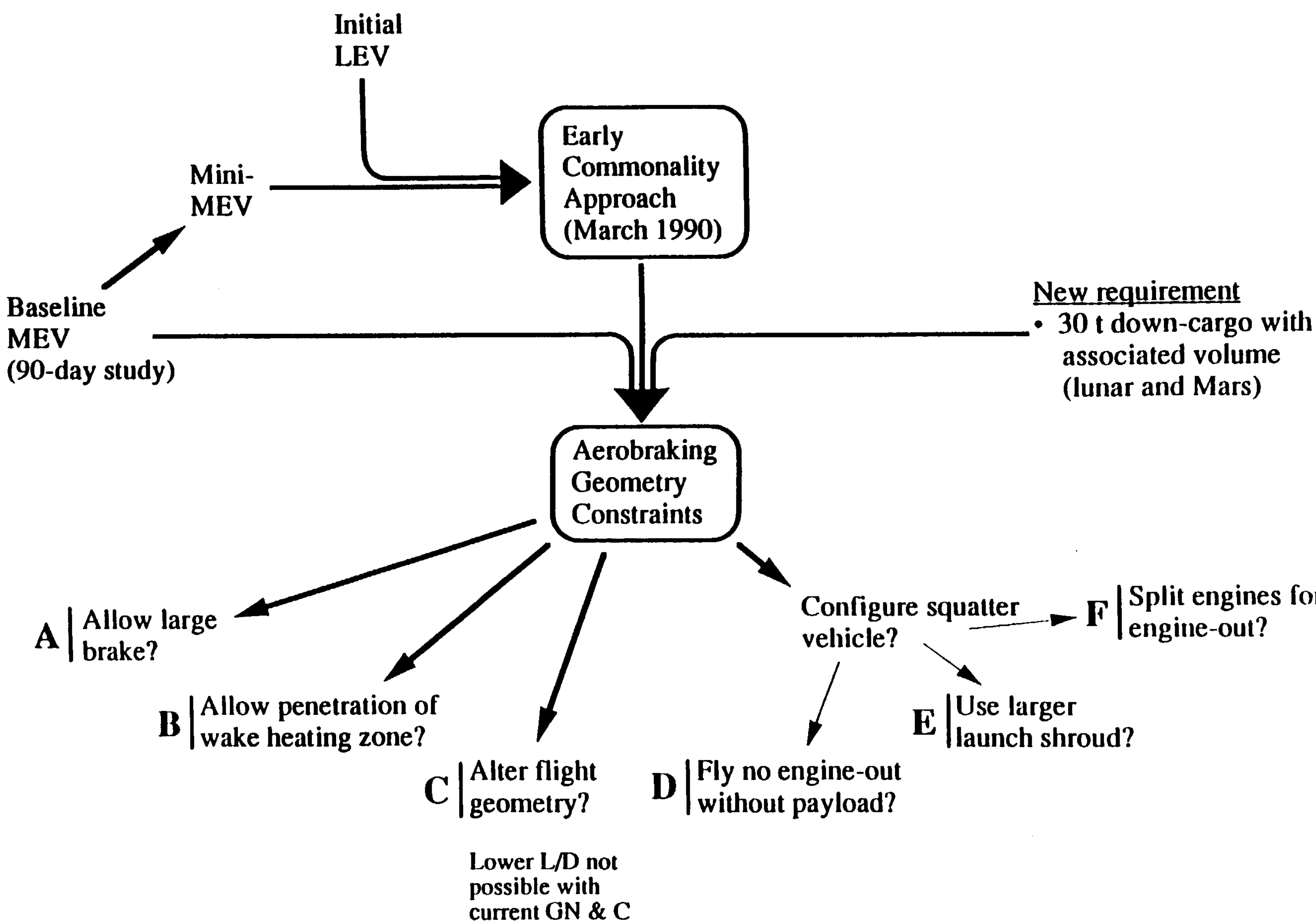
"Performing subsystem" level: **Use system assemblies or components *from the same production line* in different functional elements** (e.g. standardized processors, ECLSS units, hatches and engines)

- **Off-the-shelf common subsystem prototypes** (allows different vendors to develop systems and interfaces to the same specifications)
- **Multiple sources meeting identical specs** (retains late program resiliency; allows procurement competition)

Full-size Lunar/Mars Excursion Vehicle Commonality

Outlined is the history of STCAEM's LEV/MEV commonality assessment. From the baseline MEV (90-day study), a common mini-MEV/preliminary-LEV concept was assessed, and an early commonality approach was studied. A new Level II 30 t down-cargo requirement at both planets complicates that initial commonality picture, because it brings the vehicle capacities into a range where inescapable differences between lunar and Mars lander Δv, and between all-propulsive and aerobraked descent, are difficult to reconcile. Design for aerobraking is an overwhelming configuration driver; 6 solution methods are suggested.

Full-size Lunar/Mars Excursion Vehicle Commonality



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Space Transfer Design for Commonality

This matrix summarizes the required design features, in several subsystem categories, for an evolutionary LTV/LEV system, the mini-MEV, and a small MTV to match which could be applied to achieve the 2010, "easy" opposition opportunity. The right column collects similar requirements into a set of design parameters which would encourage direct commonality between lunar and early Mars missions, based on cryogenic propulsion and aerobraking technology. Aerobrake structures tend not to scale well geometrically, quite apart from the requirement to tailor the structural weight of each to its design payload (so that its mass-reducing benefits can be realized); however, for the purpose of this investigation we have chosen an identical size to work with for both Earth return from the Moon and Mars landing.

Space Transfer Design for Commonality

	LTV	LEV	Mini MEV	2010 MTV	Design Case
Crew cab	- 6 crew, 9 d - rad shelter - partial closure	- 6 crew, 3 d - contam. ctrl. - open ECLSS	3 crew, 10 d - contam. ctrl. - open ECLSS	- 3 crew, 1020 d - rad shelter - closed ECLSS	- 4.4m diameter system - modular lengths - optional rad shelter
Avionics	- Deep space - Orbital - R & D	- Terrain - Orbital - Rend & Dock	- Aerobraking - Terrain - Orbital - Rend & Dock	- Deep space - Aerobraking - Orbital - Rend & Dock	- Modular avionics
Propellant tanks	110t load, cryo	25t load, cryo	- 21t load, cryo - vacuum jackets	140t load, cryo	- 25t tankset, vacuum jacket upgrade - 110t tankset
Engines	3 30klbf 1 engine-out	3 30klbf 1 engine-out	3 30klbf 2 engine-out asc.	3 30klbf 1 engine-out	- 30klbf engine - structure & plumbing
Structure	- strut frame - 7.6m pieces integr. on orbit	- strut frame - 7.6m compat. launched intact	- strut frame - 7.6m compat. launched intact	- strut frame - 7.6m pieces integr. on orbit	- common approach
Landing legs	-------------	22m footprint	16m footprint	-------------	- common approach
Aerobrake	- L/D = 0.2 - 23m length - engine port	-------------	- L/D = 0.5 - 26m length - engine port	L/D = 0.5	- 0.5 L/D shape - 26m length - opt. engine section
Payload accommod.	- mission unique - transferable	- mission unique - transferable	- 1t total - rover & science	- comsats - transit science	- lunar system pallet - Mars unique attach.

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Commonality Assessment

Element	potentially common with...	...but with specific exceptions
MEV crew module	LEV / LTV crew modules	Space dormancy time
MAV	LEV / LTV engines, ACS, avionics	Avionics (rendezvous GN&C for elliptic orbits)
MEV descent stage	LEV engines & avionics	Avionics (descent GN&C with aerobraking)
MEV aerobrake	MTV aerobrake	Reduced structural gauges for lighter payload; engine ports; possibly different TPS
MTV crew module	Surface modules, ECLSS	Airlock / dustlock design
MTV / TEIS	LTV	Tank & structure arrangement may be different
MTV aerobrake	LTV aerobrake?	Probably evolution rather than commonality (different size, heating rate, payload mass, L/D)
TMIS	HLLV 3rd stage (Shuttle-Z); LTV engines in low-thrust options	Needs MLI for cryo storage; longer burn time

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Common Lunar/Mars Lander Mass Statement
Cargo and Manned Versions

Shown are mass-based designs for the current LEV/MEV commonality analysis. A single-stage vehicle was baselined into the requirements, and the primary criterion was to keep the complete propulsion stage identical for all vehicles. Vehicle inert weight is a function of several requirements, foremost being the requirement to provide a minimum vehicle T/W ratio of about 1.6 throughout both landing and ascent phases, with an engine-out margin. Other important parameters include: the stage structural frame and landing leg weight, both of which are a function of the load that each must support; and the number of tanks, and their size, selected for both the MPS and RCS propellant loads (which include boiloff allowances for surface stay time in the case of the piloted sorites). For this investigation, the common inert stage was driven by the piloted Mars lander mission with a 30 d surface stay time. With one hydrogen tank, four small oxygen tanks (total propellant capacity of 31 t), three 30 klbf advanced cryogenic engines (I_{sp} = 475 s), insulation, meteor shields, VCS, propellant lines, frame structure, landing legs and mass growth allowance all included, a common 7.4 t inert stage design resulted.

A one-year staytime on the Mars surface introduces such large boiloff losses that vehicle inerts get much heavier. This case was therefore not baselined into the commonality requirements. (Surface refrigeration may be a better trade than a permanent vehicle inert mass penalty.)

The single inerts design is seen to accommodate both lunar and Mars requirements. In fact, more cargo can be landed on the Moon -- limited by allowable thrust-to-weight -- in both crew and cargo modes if the standard tanks are not offloaded as they would be for the nominal 30 t cargo. The aerobrake mass was assumed fixed, and does not reflect the penalty of a larger aerobrake required for some of the configuration options identified.

Common Mars/Lunar Lander

The objective here was to design a common Mars/Lunar lander that could operate either as an unmanned cargo carrier, or as a piloted vehicle carrying personnel and cargo to and from the surface. The primary design criterion was to keep the complete propulsion stage identical for all vehicles. The vehicle inert weight is a function of several requirements, foremost among these being the propulsion requirement to provide a minimum vehicle T/W ratio of approximately 1.6 throughout the landing and ascending phases with an engine out margin. Other important parameters include the stage structural frame and landing leg weight, both of which are a function of the load that each must support. Inert weight is also a function of the number of tanks and their size as selected for both the MPS and RCS propellant loads which include boiloff allowances for surface stay time in the case of the piloted sorties. For this preliminary design, the decision was made to size the common inert stage based on the piloted Mars lander mission with a 30 day surface stay time. A total cryogenic propellant load of 31 tons was necessary for this case. To provide a T/W of 1.6, three 30k lbf advanced cryogenic engines (Isp=475) were selected. One large fuel tank, and small 4 oxidizer tanks were selected and sized to hold the 31 ton maximum. Adding tank insulation, meteor shields, VCS, propellant line wts, frame structure, landing legs and mass growth allowance to the engine/tank set produced a total stage inert weight of 7.4 t. The following comments will serve as a brief description for the 6 vehicles given summary weight statements on the following chart.

Column (1): Mars cargo: 7.5 t aeroshell & 30 t cargo to the surface requires only 8.1 tons of prop (1/4 of tank vol) .

Column (2): Mars piloted with 1 year stay: 7.5 t aeroshell, 4.8 t ascent cab (crew of 6), 500 kg of cargo, and 1 year boiloff allowance requires 37 tons total desc & asc prop, which is *above* the common stage tank set load capacity, thus *excluding it from the group sharing the common propulsion stage*.

Column (3): Mars piloted with 30 day stay: requires 31 t prop; selected as the capacity for sizing the tanks for the common stage.

Column (4): Lunar cargo: 30 tons to the surface implies off loaded tanks: only 21.5 tons propellant required (2/3's full)

Column (5): Lunar cargo: 45 tons to the surface are possible if the tanks are at the 31 ton maximum capacity.

Column (6) Piloted Lunar: ascent cab with 23 tons of surface cargo are possible if tanks are filled to their 31 ton capacity.

For the Mars missions a 7.5 t aeroshell decelerates the craft for the majority of the descent dV. Part way through this aerobraking phase the nozzles of the 3 engines are extended through the brake doors. Supplemental braking is provided by these engines until the brake drops off from its own weight, after which the engines alone provide terminal descent to hover and final touchdown. The Lunar cargo case is identical to the Mars cargo case excepting the use of the aeroshell. For the manned sorties the lander functions as a single stage descent/ascent vehicle. The entire vehicle ascends to orbit leaving behind only the surface cargo for the piloted lunar case, while leaving the landing legs as well for the piloted Mars case.

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Common Lunar/Mars Lander Mass Statement Cargo and Manned Versions

Mars desc propul dV: 773, Asc dV: 5319, Lunar asc dV: 2100, desc dV: 2000, all cryo prop Isp=475
Piloted vehicles (single stg) *Mars: aeroshell, cargo & landing legs left on surface, Lunar: cargo left on surf*
Cargo landers (unmanned) *Mars: aerocapture, aerobrake & propulsive descent, Lunar: all propulsive*
RCS dV: Mars: desc 100 m/s, asc 35 m/s, Lunar: desc 35 m/s, asc 35 m/s

Element *Stay time*	**Mars** *Cargo* *(desc only)* *n/a*	**Mars** *Manned* *(single stg)* *365 days*	**Mars** *Manned* *(single stg)* *30 days*	**Lunar** *Cargo* *(desc only)* *n/a*	**Lunar** *Cargo* *(desc only)* *n/a*	**Lunar** *Manned* *(single stg)* *6 months*
Ascent cab	0	4847	4847	0	0	4847
Stg inerts	*7396	8656	*7396	*7396	*7396	*7396
Aeroshell	7500	7500	7500	n/a	n/a	n/a
Surf Cargo	30000	500	500	30000	**45000	**23000
Asc prop	n/a	28001	23389	n/a	n/a	7193
Desc prop	8125	9188	7713	21502	30126	24277
RCS prop	1704	2121	1854	705	988	914
Total wt	**54725**	**60810**	**53227**	**59602**	**83510**	**67627**

Manned: crew of 6, immediate transfer to surf hab module
Stay time: for Asc stg propellant boiloff calculation only
** *Maxium surface cargo load for these lunar cases when all tanks are full (not off loaded as in column 4)*
* *Commonality across vehicles is realised by using a common propulsion stg: identical engs, structure, MPS & RCS tank set differences consist in: aerobrakes, cargo load, tank prop load (off loaded tanks in columns 1 & 4)*

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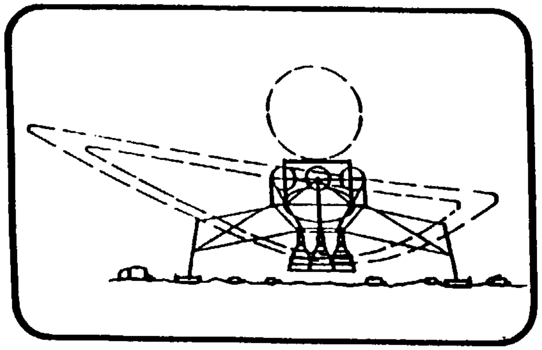
Common Lunar/Mars Vehicle Configuration Options

Shown are potential solutions, and their configuration implications, addressing five of the six options identified to reconcile aerobraking at Mars with the single-stage stack design for the LEV/MEV. None of the solutions is without problems: larger aerobrakes than baselined up until now (possibly critically limited depending on the aerobrake launch scenario); the lack of engine-out capability when the vehicle is flying emptied of payload; or a split-engine vehicle design, also a departure from baseline concepts.

Common Lunar/Mars Vehicle Configuration Options

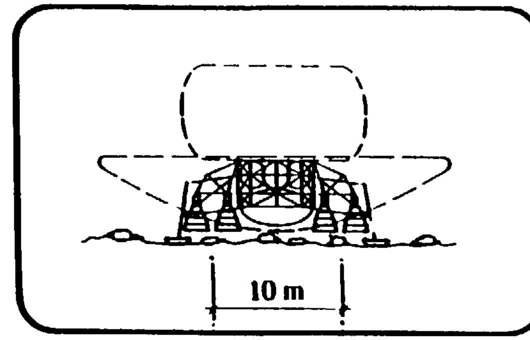


A

- 40 m aerobrake required

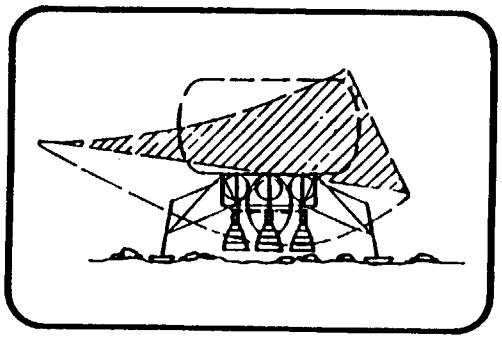
F

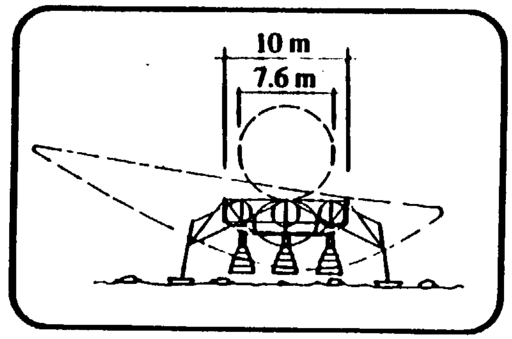
- Relax central engine cluster constraint
- Opposing shut-off for engine-out
- 4 engines required
- 30 m aerobrake required



B

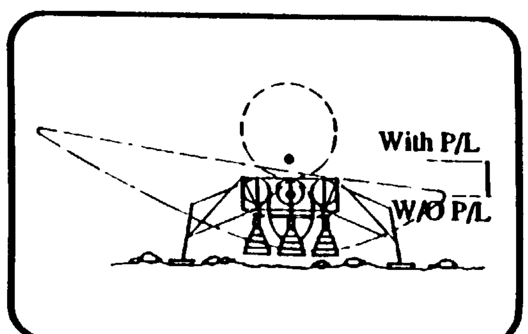
- Use topside thermal protection shroud
- 31 m aerobrake required





E

- Relax 7.6 m launch shroud constraint
- Use 10 m shroud
- Early lunar use questionable
- 35 m aerobrake required



D

- No engine-out without payload
- 35 m aerobrake required

Lunar/Mars Excursion Vehicle Commonality
Updated Preliminary Conclusions

The conclusions show that commonality is not a "*cut and dry*", simple issue to solve. Many factors affect the outcome, and are outlined below. Commonality also does not have to mean vehicles that are ***exactly*** the same because the Moon and Mars are two very different problems. Forcing lander commonality could force ***uncommonality*** of hab systems (surface and transit), transfer vehicles and aerobrakes. Commonality can, however, be seen at the subsystems level. It may be possible to develop a "kit-of-parts" which are assembled for varying missions. Commonality is an issue that will have to be pursued to a finer level to actually assess the most feasible approach.

ADVANCED CIVIL SPACE SYSTEMS

Lunar/Mars Excursion Vehicle Commonality
Updated Preliminary Conclusions

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- "Commonality" can apply to *subsystems* or to *entire vehicles*
- There are limited, sensible ranges within which each type of commonality is most feasible
- The lunar and Mars cases *are not* inherently common because of aerobraking, and may begin years apart in any case
- Cryogenic management (boiloff or refrigeration) becomes a dominant assumption for long stay times (≥ 1 year)
- Vehicle commonality at the scale of the "mini-MEV" (our early assessment) *may* indeed be feasible
- Large payload volumes have non-trivial impacts on the aerobraking cases
- Vehicle commonality in the 30 t - payload class forces new priorties, whose costs *may not* be worth it
- The common EV concept flies offloaded for lunar & Mars cargo delivery
- Full propellant tanks (31 t) *could* fly 45 t cargo to the Moon --- 23 t cargo in the lunar crew case
- Subsystem commonality (engines, mechanisms, avionics, ACS) is *probably* more appropriate for the larger class vehicles

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LEV/MEV Commonality Findings

The results of the LEV/MEV commonality investigation invite a reassessment of the function of commonality in SEI. Basic astrodynamic conditions are different enough for the Moon and Mars that optimal overall vehicles designs for each place tend in different directions.

Before a decision mandating whole-vehicle commonality is warranted, critical contextual issues must be clarified: the role of the lander in the overall transportation architecture; and the specific nature of the payloads each is likely to carry.

LEV/MEV Commonality Findings

LEV and MEV requirements are inescapably divergent

Differing gravity level (factor of 2)
- The common EV concept flies offloaded for lunar & Mars cargo delivery
- Full tanks (31 t) *could* land 45 t lunar cargo (23 t in crew case)

Aerobraking constraints (Mars only)
- Large payload volumes have non-trivial impacts on the aerobraking cases

Lunar and Mars surface mission requirements may also be divergent

Stay time is a critical parameter
- Campsite or excursion lander only?
- Boiloff or refrigeration?

Realistic lunar and Mars cargo may be substantially different

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LEV/MEV Commonality Conclusions

Shown are the basic conclusions regarding LEV/MEV commonality. Commonality at the system and subsystem levels continues to show promise, but whole-vehicle commonality appears to result in really efficient satisfaction of requirements only at the scale of the mini-MEV analyzed earlier. Driving for vehicle commonality in the "full-size" Mars lander range does not result in so good a match.

LEV/MEV Commonality Conclusions

- Vehicle commonality at the scale of the "mini-MEV" *may* indeed be feasible, as suggested earlier
- LEV/MEV commonality in the 30 t payload class forces new configuration priorities, whose costs *may not* be worth it
- Lunar and Mars EV flights may begin years apart anyway
- System and subsystem commonality (crew cabs, engines, mechanisms, avionics modules, ACS, perhaps tanks) appears more appropriate for the larger class vehicles

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STCAEM Commonality Assessment

Shown is a matrix which encapsulates the current state of knowledge from STCAEM about the type of commonality appropriate for SEI elements "across the architecture". A high degree of potential commonality is apparent, although most often at the system level.

STCAEM Commonality Assessment

	SSF	SEI LEO Ops	Lunar Transfer	Lunar Excursion	Lunar Surface	Mars Transfer	Mars Excursion	Mars Surface
Vehicles								
Lunar Transfer	None	Subsystem	Exemplar	Element	None	Component	Component	None
Lunar Excursion	None	Subsystem	Subsystem	Exemplar	None	Technology	Subsystem	None
Mars Transfer	None	None	None	None	None	Exemplar	None	None
Mars Excursion	None	None	None	Subsystem	None	Technology	Exemplar	None
MCRV	Exemplar	Element	Subsystem	None	None	Subsystem	None	None
Vehicle Systems								
Main Propulsion	None	Subsystem	Exemplar	Subsystem	None	TBD	Subsystem	None
ACS Propulsion	None	Subsystem	Exemplar	Subsystem	None	Subsystem	Subsystem	None
Aerobrake	None	None	Component	None	None	Subsystem	Exemplar	None
Propulsive Power	None	None	Subsystem	None	Subsystem	Exemplar	None	Subsystem
Housekeeping Power	Exemplar	Subsystem	Subsystem	Component	Subsystem	Subsystem	Component	Subsystem
Communications	Technology	Component	Component	Component	Component	Exemplar	Subsystem	Subsystem
Avionics	Technology	Subsystem	Exemplar	Subsystem	Technology	Subsystem	Subsystem	Technology
Robotics	Technology	Subsystem	Subsystem	Subsystem	Subsystem	Exemplar	Subsystem	Subsystem
Thermal Control	Exemplar	Subsystem	Subsystem	Subsystem	Component	Subsystem	Component	Component
Structures	Exemplar	Component	Component	Component	Component	Subsystem	Component	Component
Crew Systems								
Small Habitat	Exemplar	Subsystem	Component	Component	Subsystem	Component	Component	Subsystem
Large Habitat	Technology	Component	Subsystem	Component	Element	Exemplar	Component	Element
Airlock	Technology	Subsystem	Subsystem	Exemplar	Subsystem	Subsystem	Subsystem	Subsystem
Suit	Technology	Subsystem	Subsystem	Exemplar	Element	Subsystem	Component	Component

Exemplar · Element · Subystem · Component · Technology · None

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B. Small/Medium/Large Scale Evolution

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V. Commonality/Evolution

B. Program Scale Evolution - The lunar programs are classified into three scales which utilize the transportation architectures in varying degrees. Each scale has a defined objective and is limited to accomplishing the goal by varying assumptions. The **small scale** scenario is a man-tended scientific outpost. Crews of four will remain on the lunar surface overnight in a small "campsite". The **medium scale** develops a permanent base for a crew of six. In addition to the scientific nature of the base, lunar oxygen will be produced by the year 2010. The most ambitious program, the **large scale**, eventually leads to permanent settlement. The base continuously grows to accommodate 18 to 24 people by the year 2025. Along with the lunar oxygen plant, industrialization is very important. Without specifying the type of the product, the industrial capabilities are equivalent to a 1 GWe Helium-3 plant.

After the program scales were established, a determination on required mass on the surface was determined along with cargo sizes and volumes for manifesting purposes. Each program scale's mass-to-the-surface requirements vary depending on the amount of ISRU, number of inhabitants, mission durations, and level of industrialization. The mass for the medium scale program in the first 10 years is almost equivalent to that needed by the large scale program. The difference is seen in the subsequent 15 years of the two programs. The medium scale program continues without major modifications to the base or number of inhabitants while the large scale program is always growing towards permanent settlement with increasing personnel.

The number of cargo flights must be based upon the mass required by the program scale. Understanding the crew rotation before scheduling the cargo flights is very important. In order to build-up the base personnel, some flights will return to the Earth without all 6 crew members on-board. The small scale program requires 13 flights of cargo and cargo in 21 years to accomplish its goals. The medium scale must have 56 flights and the large scale requires 113 flights, both within 25 years. The cargo flights are manifested as tandem LTVs since they are capable of delivering 55 t in one mission. If LOR flights with lunar oxygen are used, the number of flights increases dramatically. These flights are capable of delivering 25 t of cargo when the LEV is already in orbit around the Moon, and 15 t if the LEV is delivered from Earth.

The end product of this analysis was a fairly detailed manifesting analysis. For the small scale program, the flight rate of 13 flights in 21 years does not indicate the actual number of ETO launches and tankers that are needed to support the lunar program. For the medium scale, approximately 700 t of cargo is required on the lunar surface by 2010. After this date, the tonnage is much less. By the end of the program, 56 flights of cargo and crew have been made. At the end of the large scale program, more than 7,500 t of cargo is placed on the lunar surface. Based on the amount of material manufactured at the base, 113 flights are necessary to deliver the remaining cargo and crews. The HLLV (10 x 30 m shroud) flights are primarily based on the component masses. The aerobrake is volume-limited and thus a separate flight. The LTV is designed to be launched intact in one flight. The "campsite" and its associated LTV are also launched in a single flight. The propellant is launched separately and is transferred to the LTVs while in LEO.

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Program Scales For Transportation Architecture Analysis

Small Scale

	Lunar	Mars
Purpose	Near-side science station	Exploration Sample collection
Surface Stay	45 days Overnight campsites (LTV with shielding)	20 days Live in the landers
Beginning	2004	2015
Frequency	Flights every other year	Flights every other year (opposition or conjunction)
Growth	Pressurized rovers Flyers	Unpressurized rovers
Precursors	Lunar observer Early manned visit	2 Mars Observers Long-lived rovers

ADVANCED CIVIL SPACE SYSTEMS

Program Scales For Transportation Architecture Analysis

BOEING

Medium Scale

	Lunar	Mars
Purpose	Permanent science base	Permanent science base
Crew	6 member science	6 member science
	(additional personnel as required for base maintainence and ISRU processes)	
Rotation	Yearly	Each opposition opportunity (possible unoccupied periods)
Beginning	2000 (no SSF support)	2010
Access	Global	Regional Pressurized rovers
CELSS	Yes	Yes (possible problems due to unoccupied periods)
Precursors	1 manned visit	>3 manned visits
Other	Nuclear surface power In-situ science Science facilities >10 km from habitat site	Robotic cargo flights

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Program Scales For Transportation Architecture Analysis

Large Scale
(builds on Medium Scale scenario)

	Lunar	Mars
Purpose	Settlement and industrialization	Settlement
Rotation	2 Years	Change-out opportunity 3 to 5 years
Beginning	2000 (no SSF support)	First Mission TBD (12 people each flight starting in 2020)
Access	Global	Global by 2025 (if possible)
Manufacturing	Equivalent to He production (>1 GWe) >80% in-situ facility production	80% in-situ facility production

ADVANCED CIVIL SPACE SYSTEMS

BOEING

Representative Lunar Base Applications

	Small	Medium	Large
Purpose	Science Station	Permanent Base	Settlement Industrialization
Surface Stay	45 days Overnight Campsites	Permanent Yearly crew rotation	Permanent 2-year rotation
Personnel	4 per visit	6 member science (additional personnel as req'd for base maintainence and ISRU processes)	TBD
Beginning	2004	2000 (no SSF support)	2000
Access	Regional	Global	Global
CELSS	No	Yes	Yes
Surface Power	Solar / RFC	Nuclear	Nuclear
Manufacturing	No	Experimental only except for lunar/Mars oxygen production	Equivalent to He production (>1 GWe) >80% in-situ facility production Solar powered facilities
Precursors	Lunar Observer Early manned visit	1 manned visit	Medium scenario

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Lunar Mission - Small Scale

ASTRONOMY

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Small Lunar Automated Telescope	400	1.5	TBD	Automated
* Lunar Gravity Wave Detector	5000	20	TBD	Set up components
* Crater Telescope	500	2	0.5/Element	2/EVA;2 crew/telescope; set up: 2hrs/telescope; 2 crew; Maintenance: 1 EVA/yr
* UV-Visible Interferometer (4 Elements)	4000	30	0.5/Element	
* Sub-millimeter (IR) Array (1 Element)	6000	50	0.5/Element	
* Lunar Optical Interferometer				Construction of Moon facilities; continuous ground operation
Initial 3 Telescopes	4300	130	2.5	
Final 12 Telescopes (by mission end)	16000	470	8.5	
TOTAL	15900	103.5	TBD	

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Lunar Mission - Small Scale

LIFE SCIENCES

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Exobiology (Biostack, Asceptic Samplers)	50	TBD	TBD	Included in geology EVA
* Biological Sample Management Facility	125	0.5	0.25	TBD
* Bioregenerative Life Support Facility	300	0.5	0.3	TBD
* Biolab - ESA	272	0.9	1.13	TBD
* Life Science Equip. NASDA	590	1.75	1.95	TBD
* CSA Life Science	42	0.31	0.551	TBD

Lunar Mission - Small Scale

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Lunar Astronaut Field Package	300	1	On-board	Direct Operation of some equipment
* Lunar Environment Station	100	TBD	On-board	Some station deployment
* Lunar Sample Return Containers (10 Sets)	200			
(Delivery)		0.3	0	Fill and stow for return
(Return)		1.2	0	
* Lunar Penetrator with Descent Imaging	100	1	On-board	Command generation Data relay
* Laser Ranging Retro-reflector	50	TBD	0.1	Emplaced during geology EVA Maintenance: 4 hrs/yr
* Particles and Field Stations (2 stations and 1 Particle detector)	1000	TBD	RTG	
TOTAL	4200	TBD	TBD	

Lunar Mission - Small Scale

TRANSPORTATION REQUIREMENTS

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Campsite	TBD	TBD	TBD	TBD
* LEV	TBD	TBD		
* Rovers				
Presssurized	TBD	TBD		
Unpressurized	TBD	TBD		
Robotic	800	16		
* Flyers				
Pressurized (25 t. payload)	55 t.	Mini MEV		
Unpressurized (t. payload)	10 t.	15' x 30'		

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Lunar Mission - Medium and Large Scales

ASTRONOMY

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Small Lunar Automated Telescope	400	1.5	TBD	Automated
* Lunar Gravity Wave Detector	5000	20	TBD	Set up components
* Crater Telescope	500	2	0.5/Element	2/EVA;2 crew/telescope; set up: 2hrs/telescope; 2 crew; Maintenance: 1 EVA/yr
* UV-Visible Interferometer (4 Elements)	4000	30	0.5/Element	
* Sub-millimeter (IR) Array (1 Element)	6000	50	0.5/Element	

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Lunar Mission - Medium and Large Scales

ASTRONOMY

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Large Optical Telescope				
4 - m Configuration	15000	300	1.5	All of these listed here require facility construction, continuous ground control, and maintenance.
16 - m Configuration	42000	TBD	5	
* Lunar Transit Telescope				
0.8 - m Aperture	1000	TBD	0.35	
1.8 - m Aperture	1250	8	0.35	
3.5 - m Aperture	4800	TBD	0.35	Each system is large, with different baseline lenghts.
* Optical Interferometer				
Initial 3 Telescopes	4300	130	2.5	
Final 12 Telescopes	16000	470	8.5	
* Nearside VLF Imaging Array	500	TBD	TBD	
* Farside VLF Interferometer	1000	TBD	1	
* Sub Millimeter Interferometer	12000	700	20	
TOTAL	111750	TBD	TBD	

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Lunar Mission - Medium and Large Scales

LIFE SCIENCES

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* 1.8-m Centrifuge Facility	1480	2	2.4	Included in geology EVA
* Habitat Holding System	500	1	1.9	TBD
* Biologincal Sample Management Facility	125	0.5	0.25	
* Bioinstrumentaltion & Physiologicl Monitoring Fac.	750	2.5	0.8	TBD
* Analytical Instruments Fac.	250	0.75	0.65	
* Life Science Lab Support Equipment Facility	325	1	0.6	TBD
* Centralized Life Science Computer Fac.	50	0.25	0.4	TBD
* Bioregenerative Life Support Facility	300	0.5	0.5	
* Exobiology Facility	250	0.5	0.5	

Lunar Mission - Medium and Large Scales

LIFE SCIENCES

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Commercial Life Sciences Facility	300	1	1	
* ESA equipment	580	1.9	3	
* NASDA Equipment	2713	8.825	10.7	
* CSA Equipment	172	1.64	2.451	
TOTAL	7695	47.115	32.351	

ADVANCED CIVIL SPACE SYSTEMS

Lunar Mission - Medium and Large Scales

BOEING

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Lunar Astronaut Field Package	300	1	On-board	Direct Operation of some equipment
* Lunar Environment Station	100	TBD	On-board	Some station deployment
* Lunar Sample Return Containers (10 Sets)	200			Fill and stow for return
(Delivery)		0.3	0	
(Return)		1.2	0	
* Lunar Penetrator with Descent Imaging	100	1	On-board	Command generation Data relay
* Laser Ranging Retro-reflector	50	TBD	0.1	Emplaced during geology EVA
* Particles and Field Stations (2 stations and 1 Particle detector)	1000	TBD	RTG	Maintenance: 4 hrs/yr

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Lunar Mission - Medium and Large Scales

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)
* Geoscience Laboratory Intstruments			
Emplacement Phase	40	0.04	0.056
Consolidation Phase	24	0.05	0.096
* Lunar 10 - m Drill	200	0.1	5.02
* Lunar Deep Drill	20000	200	500
TOTAL	24464	TBD	505.272
* Analytical Science Lab. Equipment	1000	TBD	10
* Geologic Exploration Equipment	100	TBD	TBD
* Drilling Equipment	1000	TBD	10/3 days
* Geophysical Stations (6 @ 100 kg)	600	TBD	RTG
* Portable Geophysical Package	100	TBD	TBD

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Lunar Mission - Medium and Large Scales

PLANETARY APPLICATIONS

Medium Scale:

Lunar LOX

- Baseline: *Robotic Lunar Surface Operations* (begin on p. 73)
- Production of 100 tons/year Need to determine production rate as to scale plant
- Time AFL to habitability: 1.5 years

Large Scale:

1 GWe He Power Plant

- Base on *Report of NASA Lunar Energy Enterprise Case Study Task Force*
- Determine amount of He required
- Etc.

Lunar Mission - Medium and Large Scales

TRANSPORTATION REQUIREMENTS

	Mass (kg)	Volume (m^3)	Power (kW)
* LTV			
* LEV			
*Habitation Modules	TBD	TBD	TBD
*Rovers	20 t + equip.		
Pressurized			
Unpressurized			
*Flyer			
Unpressurized (t payload)	10000	15'x30'	
Pressurized (25 t payload)	55000	mini MEV	

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Lunar Flights - Small Scale

ASSUMPTIONS:

- Flights approximately every other year.
- Use tandem LTVs - LEV hasn't been developed.
- Scientific outpost
- Overnight stays in a campsite
- Crew size of 4

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Lunar Flights - Small Scale

Flight Number	Year	Mission Description
1	2004	Crew of 4 - 4 to 6 days on Lunar Surface - Direct flight using tandem LTVs
2	2005	Campsite delivery.
3	2006	Crew of 4 - First overnight stay; - Tandem LTV fight to Lunar Surface; - Science based mission.
4	2008	Cargo delivery.
5	2009	Crew of 4 - Overnight stay; - Science based mission.
6	2011	Crew of 4 - Overnight stay
7	2013	Crew of 4 - Overnight stay
8	2015	Science cargo delivery.
9	2017	Crew of 4 - Science based mission.
10	2019	Crew of 4 - Another overnight stay.
11	2021	Crew of 4 - Overnight stay;
12	2023	Cargo delivery.
13	2025	Crew of 4 - Overnight, science mission.

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Lunar Manifest Worksheet - Small Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2004, 2011, 2019	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake, 1 ACRV, Crew Cab, Landing Legs	2 LTVs, 1 Brake, 1 ACRV, Crew Cab, LTV Landing Legs	209 t	261 t	3	4 HLLV 1 Crew	4 Crew, 4 - 6 days for 1st mission, 40 days on later missions
2005	Campsite Delivery	LTVT Dir,exp	25 t + Camp-site	2 LTVs, 1 Brake Landing Legs	1 LTV, Campsite, LTV Landing Legs, Spare Brake (reused LTV and Brake)	206 t	290 t	3	3 HLLV	N/A
2006, 2013, 2021	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake 1 ACRV, Crew Cab, Landing Legs	Spare LTV, 1 ACRV, Landing Legs, Crew Cab (reused LTVand Brake)	213 t	267 t	3	3 HLLV 1 Crew	4 Crew, 40 days on LS
2008, 2015, 2023	Science Cargo	LTVT Dir,exp	55 t	2 LTVs, 1 Brake Landing Legs	Science Cargo, 1 LTV, Landing Legs (reused LTV and Brake)	206 t	290 t	3	3 HLLV	N/A
2009, 2017, 2025	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake 1 ACRV, Crew Cab, Landing Legs	1 LTV, 1 ACRV, Landing Legs, Crew Cab (reused LTV and refurbished Brake)	213 t	267 t	3	3 HLLV 1 Crew	4 Crew, 40 days on LS

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Lunar Flights - Medium Scale

ASSUMPTIONS:

- One crew flight per year.
- Cargo flight rates will vary.
- Flights begin in 2000.
- Initially no Space Station support.
- By 2010, approximately 700 tonnes are needed on the lunar surface. This includes the LLOX plant and base construction materials.
- A campsite will be used to house the crew until the base is completed.
- Crew flights initially have 4 crew members until 2010. Crew size increases to 6.
- Lunar Oxygen will be used for ascent and descent of the LEV beginning in 2010.

Lunar Flights - Medium Scale

Flight Number	Year	Mission Description
1	2000	Crew Sortie
2	2000	Campsite delivery.
3	2001	Deliver material for base construction.
4	2001	Crew of 4 - Base construction and inspection of robotic work; -Stay in Campsite.
5	2002	Deliver construction material.
6	2002	Crew of 4 - Mission similar to Flight 4.
7	2003	Cargo delivery.
8	2003	Crew mission.
9	2004	Delivery of LLOX plant construction material.
10	2004	Crew of 4 - Inspection and construction of LLOX plant and lunar base.
11	2005	Deliver materials for LLOX plant and habitats
12	2005	First crew of 6.
13	2006	Cargo delivery of construction materials.
14	2006	Crew of 6 - Continues aconstruction.

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Lunar Flights - Medium Scale

Flight Number	Year	Mission Description
15-16	2007	Cargo deliveries of materials and supplies.
18	2007	Crew of 6 - 1 year stay.
19-20	2008	Deliver material for base construction.
21	2008	Crew of 6 - Base construction and inspection of robotic work; -Stay at the base.
22-23	2009	Deliver construction material.
24	2009	Crew of 6 - Mission similar to Flight 21.
25	2010	Cargo delivery - Uses LLOX
26	2010	First crew mission to use LLOX.

In the period of **2010 - 2025**, there will be **15** crew missions which use lunar oxygen as the oxidizer. Also during this time, a minimum of **15** LLOX cargo flights have been manifested. This totals to **56** flights during the program. It is possible to have more flights, but these have not been accounted for. If large payloads are necessary, tandem LTVs should be used in place of the LLOX flights.

Crew Rotation - Medium Scale

Launch	Return	Stay	Base Crew
00 00 00 00	00 00 00 00	n/a	0
01 01 01 01	01 01 01 01	n/a	0
02 02 02 02	02 02 02 02	n/a	0
03 03 03 03	03 03 03 03	n/a	0
04 04 04 04	04 04 04 04	n/a	0
05 05 05 05 05 05	05 05 05 05 05 05	n/a	0
06 06 06 06 06 06	06 06 06 06 06 06	n/a	0
07 07 07 07 07 07	07 07 07	07 07 07	3
08 08 08 08 08 08	07 07 07 08 08 08	08 08 08	3
09 09 09 09 09 09	08 08 08	09 09 09 09 09 09	6
10 10 10 10 10 10	09 09 09 09 09 09	10 10 10 10 10 10	6
11 11 11 11 11 11	10 10 10 10 10 10	11 11 11 11 11 11	6
12 12 12 12 12 12	11 11 11 11 11 11	12 12 12 12 12 12	6
13 13 13 13 13 13	12 12 12 12 12 12	13 13 13 13 13 13	6

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

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Crew Rotation - Medium Scale

Launch	Return	Stay	Base Crew
14 14 14 14 14 14	13 13 13 13 13 13	14 14 14 14 14 14	6
15 15 15 15 15 15	14 14 14 14 14 14	15 15 15 15 15 15	6
16 16 16 16 16 16	15 15 15 15 15 15	16 16 16 16 16 16	6
17 17 17 17 17 17	16 16 16 16 16 16	17 17 17 17 17 17	6
18 18 18 18 18 18	17 17 17 17 17 17	18 18 18 18 18 18	6
19 19 19 19 19 19	18 18 18 18 18 18	19 19 19 19 19 19	6
20 20 20 20 20 20	19 19 19 19 19 19	20 20 20 20 20 20	6
21 21 21 21 21 21	20 20 20 20 20 20	21 21 21 21 21 21	6
22 22 22 22 22 22	21 21 21 21 21 21	22 22 22 22 22 22	6
23 23 23 23 23 23	22 22 22 22 22 22	23 23 23 23 23 23	6
24 24 24 24 24 24	23 23 23 23 23 23	24 24 24 24 24 24	6
25 25 25 25 25 25	24 24 24 24 24 24	25 25 25 25 25 25	6

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

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Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2000	Crew Sortie	LTVT Dir/fo	15 t	2 LTVs, 1 Brake, Landing Legs	2 LTVs, 1 Brake, Landing Legs	202 t	255 t	3	4 HLLV 1 Crew	4 Crew, 4 - 6 days on LS
2001	Crew	LTVT Dir/fo	15 t	2 LTVs, 1 Brake, Landing Legs	1LTV, 1 Brake, Landing Legs	220 t	277 t	3	3 HLLV 1 Crew	4 Crew, 40 days on LS
2002	Crew	LOR	3.5 t	1 LTV, 1 LEV, 1 LTVCM, 1 LEVCM, Aerobrake	1 LTV, 1 LEV, 1 LEVCM, 1 LTVCM, Aerobrake	132 t	174 t	2	4 HLLV 1 Crew	4 Crew, 40 days on LS
2003-2004	Crew	LOR	10 t	1 LTV, 1 LEV, 1 LTVCM, 1 LEVCM, Aerobrake	None (reuse everything)	134 t	170 t	2	1 Crew (Cargo in STS flight)	4 Crew, 6 months on LS
2005-2006	Crew	LOR	10 t	1 LTV, 1 LEV, 1 LTVCM, 1 LEVCM, Aerobrake	None (reuse everything)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 6 month stay

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Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2007	Crew	LOR	3 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	132 t	173 t	2	3 HLLV 1 Crew	6 Crew, 1 year on LS
2008, 2009	Crew	LOR	~~11~~ t 10	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components)	~~137~~ t 134	~~174~~ t 171	2	1 HLLV 1 Crew	6 Crew, 1 year on LS
2010, 2015, 2020, 2025	Crew	LOR	6 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LTV, LTVCM, LEV, LEVCM, Aerobrake	116 t	159 t	2	3 HLLV 1 Crew	6 Crew, 1 year on LS
2011, 2016, 2021	Crew	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components)	94 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 1 year on LS
2012, 2017, 2022	Crew	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components)	94 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 1 year on LS

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Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2013, 2018, 2023	Crew	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components and Brake)	94 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 1 year stay
2000	Campsite Delivery	LTVT Dir/fo	24 t + Campsite	2 LTVs, 1 Brake, Landing Legs	2 LTVs, Landing Legs, Brake, Campsite	206 t	290 t	3	4 HLLV	N/A
2001-2009	Cargo	LTVT Dir/fo	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs (reuse 1 LTV and Brake)	206 t	290 t	2	2 HLLV	N/A
2007-2009	Cargo	LTVT Dir/fo	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs (reuse 1 LTV and Brake)	206 t	290 t	2	2 HLLV	N/A
2010	Cargo	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM	1 LTV, 1 LEV, Aerobrake	99 t	135 t	1	3 HLLV	N/A
2011	Cargo	LOR	25 t	1 LTV, 1 Brake, 1 LTVCM	None (reuse everything)	81 t	120 t	1	1 HLLV	N/A

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Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population

After the initial 700 tonnes of material have been delivered to the lunar surface, cargo flights may use Lunar LOX as the oxidizer. If large payloads are necessary, the direct, tandem LTV flights can be used. Otherwise, the cargo missions will be similar to those of 2010 and 2011, depending on the LTV and LEV requirements.

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Lunar Flights - Large Scale

ASSUMPTIONS:

- Flights begin in 2000.
- Minimum of one crew flight per year.
- Rate of cargo flights vary
- Beginning 2015, 80% of habitat facilities are derived from in-situ resources.
- Logistics:
  - 5 kg/person-day and a 2 year stay by 2010
  - 3 kg/person-day and a 3 year stay by 2015
  - 2 kg/person-day and a 4 year stay by 2020
- For industrialization, assume
  - Production plant mass = 2 * (annual production)
  - Lunar oxygen plant mass = 1.2 * (annual production)
- Additional crew flights are not required to remain for long durations on the lunar surface. These missions may be "visitations" for scientific purposes.

Mass Required On the Lunar Surface

By 2010	(both Medium and Large Scale Programs)	
	Life Support Hardware	31 t
	LLOX Plant Materials	250 t
	Habitat Structures	250 t
	Additional Mass	169 t
	Total	**700 t**
By 2015	(Large Scale Program only)	
	Life Support Hardware	100 t
	LLOX Plant	250 t
	Habitat Structures	750 t
	Helium-3 Plant Materials	500 t
	Additional Mass	400 t
	Total	**2000 t**
By 2025	(Large Scale Program only)	
	Life Support Hardware	250 t
	LLOX Plant	300 t
	Habitat Structures	1000 t
	Helium-3 Plant Materials	5000 t
	Additional Mass	950 t
	Total	**7500 t**

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Cargo Delivery Requirements - Large Scale

By 2010

- 700 tonnes of material on the lunar surface.
- Use tandem LTVs capable of 55 ton deliveries.
- Requires 12 cargo flights and 1 Campsite delivery.

By 2015

- Additional 2300 tonnes of material are needed
- In-situ resource utilization
  - -- 40% of the additonal 500 tonnes for the habitat modules
  - -- Assume 20% of the Helium-3 plant mass.
- Must deliver 1000 tonnes from the Earth.
- Requires 19 tandem LTV cargo flights.

By 2025

- 4500 tonnes of addtional material are needed.
- In-situ resource utilization
  - -- 50% of the new life support system hardware
  - -- 80% of the remaining habitat mass
  - -- 50% of the additional Helium-3 plant mass
- Must deliver 2975 tonnes from the Earth.
- Requires 54 tandem LTV cargo flights
- May use additional LLOX cargo flights (these have not been manifested)

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Lunar Flights - Large Scale

Flight Number	Year	Mission Description
1	2000	Crew Sortie
2	2000	Campsite delivery.
3	2001	Deliver material for base construction.
4	2001	Crew of 4 - Base construction and inspection of robotic work
5	2002	Crew of 4 - Base construction and inspection of robotic work; -Stay in Campsite.
6	2002	Cargo delivery for LLOX plant construction and base build-up.
7	2003	Delivery of LLOX plant and base construction material.
8	2003	Crew of 4 - Inspection and construction of LLOX plant.
9	2004	Deliver materials for production plant, LLOX plant, and base.
10	2004	Crew of 4 - First to stay at the base (stay time of 6 months) - Continues construction and base build-up.
11	2005	Cargo delivery of construction materials and crew supplies
12	2005	Crew of 6 - 3 will remain 6 months while 3 will remain for 1 year.
13	2006	Deliver material for base construction.
14	2006	Crew of 6 - All 6 will remain on surface; - Return crew from previous flight.

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Lunar Flights - Large Scale

Flight Number	Year	Mission Description
15	2007	Deliver construction materials and supplies.
16	2007	Crew of 6 - Replaces crew at the base for 1 year.
17	2007	Delivery of materials and supplies.
18-19	2008	Deliver materials for construction.
20	2008	Crew of 6 - Change-out of base personnel
21-22	2009	Cargo delivery.
23	2009	Crew of 6 - Replaces base personnel; - Stay time of 1 year.
24-27	2010	Cargo delivery.
28	2010	Crew of 6 - Change-out of base crew.
29-32	2011	Deliver material for base expansion, production plant
33	2011	Crew of 6 - Crew change-out at the base; - Only 3 from previous crew returns to Earth. - Base now houses 9 crew.
34-37	2012	Deliver material for base expansion.
38	2012	Crew of 6 - Replace some of base personnel; - 9 remain on surface
39-42	2013	Cargo delivery
43	2013	Crew of 6

Lunar Flights - Large Scale

Flight Number	Year	Mission Description
44-46	2014	Cargo delivery - plant construction and base build-up.
47	2014	Crew of 6
48-52	2015	Cargo delivery
53	2015	Crew of 6
54	2016	Crew of 6 - Change-out of base personnel
55-59	2016	Cargo delivery.
60-65	2017	Cargo delivery.
66	2017	Crew of 6 - Replaces base personnel;
67-72	2018	Cargo delivery.
73	2018	Crew of 6 - Change-out of base crew.
74-79	2019	Deliver material for base expansion.
80	2019	Crew of 6 - Replace some of base personnel;
81-86	2020	Cargo delivery
87	2020	Crew of 6
88-92	2021	Cargo delivery - plant construction and base build-up.
93	2021	Crew of 6

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Lunar Flights - Large Scale

Flight Number	Year	Mission Description
94-98	2022	Cargo delivery
99	2022	Crew of 6
100	2023	Crew of 6 - Change-out of base personnel
101-105	2023	Cargo delivery.
106-110	2024	Cargo delivery.
111	2024	Crew of 6 - Change-out of base personnel.
112	2025	Crew of 6 - Replaces base personnel.
113	2025	Cargo delivery.

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Crew Rotation - Large Scale

Launch	Return	Stay	Base Crew
00 00 00 00	00 00 00 00	n/a	0
01 01 01 01	01 01 01 01	n/a	0
02 02 02 02	02 02 02 02	n/a	0
03 03 03 03	03 03 03 03	n/a	0
04 04 04 04	04 04 04 04	n/a	0
05 05 05 05 05 05	05 05 05 05 05 05	n/a	0
06 06 06 06 06 06	06 06 06 06 06 06	n/a	0
07 07 07 07 07 07	07 07 07	07 07 07	3
08 08 08 08 08 08	07 07 07 08 08 08	08 08 08	3
09 09 09 09 09 09	08 08 08	09 09 09 09 09 09	6
10 10 10 10 10 10	09 09 09 09 09 09	10 10 10 10 10 10	6
11 11 11 11 11 11	10 10 10	10 10 10 11 11 11 11 11 11	9
12 12 12 12 12 12	10 10 10	11 11 11 11 11 11 12 12 12 12 12 12	12
13 13 13 13 13 13	11 11 11 11 11 11	12 12 12 12 12 12 13 13 13 13 13 13	12

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

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Crew Rotation - Large Scale

Launch	Return	Stay	Base Crew
14 14 14 14 14 14	12 12 12 12 12 12	13 13 13 13 13 13 14 14 14 14 14 14	12
15 15 15 15 15 15	13 13 13 13 13 13	14 14 14 14 14 14 15 15 15 15 15 15	12
16 16 16 16 16 16	14 14 14 14 14 14	15 15 15 15 15 15 16 16 16 16 16 16	12
17 17 17 17 17 17	15 15 15	15 15 15 16 16 16 16 16 16 17 17 17 17 17 17	15
18 18 18 18 18 18	15 15 15	16 16 16 16 16 16 17 17 17 17 17 17 18 18 18 18 18 18	18
19 19 19 19 19 19	16 16 16 16 16 16	17 17 17 17 17 17 18 18 18 18 18 18 19 19 19 19 19 19	18

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

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Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2000	Sortie	LTVT Dir/fo	10 t	2 LTVs, 1 Brake, Landing Legs, Excursion Cab, ACRV	2 LTVs, 1 Brake, Landing Legs, ACRV, Excursion Cab	209 t	261 t	3	3 HLLV 1 Crew	4 Crew, 4 - 6 days on LS
2001	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake, Landing Legs, Excursion Cab, ACRV	1 LTV, Excursion Cab, ACRV, Landing Legs (reuse 1 LTV and Brake)	209 t	261 t	3	2 HLLV 1 Crew	4 Crew, 40 days on LS
2002	Crew	LOR	3 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LEV, LTVCM, LEVCM, Brake (reuse LTV)	131 t	171 t	2	2 HLLV 1 Crew	4 Crew, 40 days on LS
2003	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	134 t	170 t	2	1 Crew (Cargo in STS flight)	4 Crew, 40 days on LS
2004	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LTV, LTVCM (reuse LEV components and Brake)	131 t	171 t	2	1 Crew (Cargo in STS flight)	4 Crew, 40 days on LS

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ADVANCED CIVIL SPACE SYSTEMS

Lunar Manifest Worksheet - Large Scale

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Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2005	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 6 months on LS
2006	Crew	LOR	3 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LEV, LEVCM, Brake (reuse LTV components)	132 t	173 t	2	2 HLLV 1 Crew	6 Crew, 6 months on LS
2007	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 3 remain at lunar base
2008	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything and refurbish brake)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 3 remain at lunar base
2009	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LTV, LTVCM (reuse LEV components and brake)	135 t	171 t	2	1 HLLV 1 Crew (Cargo in STS flight)	6 Crew, 6 remain at lunar base

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Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2010	Crew (LLOX mission)	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	93 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 6 remain at lunar base
2011	Crew (LLOX mission)	LOR	8 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LEV, LEVCM, Brake (reuse LTV components)	111 t	167 t	2	1 HLLV 1 Crew (Cargo in STS flight)	6 Crew, 9 remain at lunar base
2012-2015	Crew	LOR	15 t							
2016	Crew	LOR	8 t	Transportation requirements are the same for the remainder of the program (for LOR type missions).	These missions will be very similiar to the 2010 and 2011 missions, which is apparent by the cargo masses. The missions in 2014, 2019, and 2024 will require a new LTV and Aerobrake. This increases the number of ETO launches by 2 for those missions. The propellant loadings are not affected by this. A crew of 6 launches from the Earth in each mission. However, not all will return immediately. By 2025, the base will have a population of 18 members.					
2017-2020	Crew	LOR	15 t							
2021	Crew	LOR	8 t							
2022-2025	Crew	LOR	15 t							

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Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2000	Campsite Delivery	LTVT Dir/exp	24 t + Camp-site	2 LTVs, 1 Brake, Landing Legs	2 LTVs, 1 Brake, Landing Legs	206 t	290 t	3	4 HLLV	N/A
2001, 2006, 2009-2010, 2012-2025	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A
2002, 2007, 2009, 2011-2013, 2015-2017, 2017-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A
2003, 2007, 2010-2013, 2015-2018, 2018-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A

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Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2004, 2008, 2010-2012, 2014-2019, 2019-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A
2005, 2008, 2010, 2011, 2013-2020, 2020-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	2 LTVs, Landing Legs, 1 Brake	206 t	290 t	3	4 HLLV	N/A

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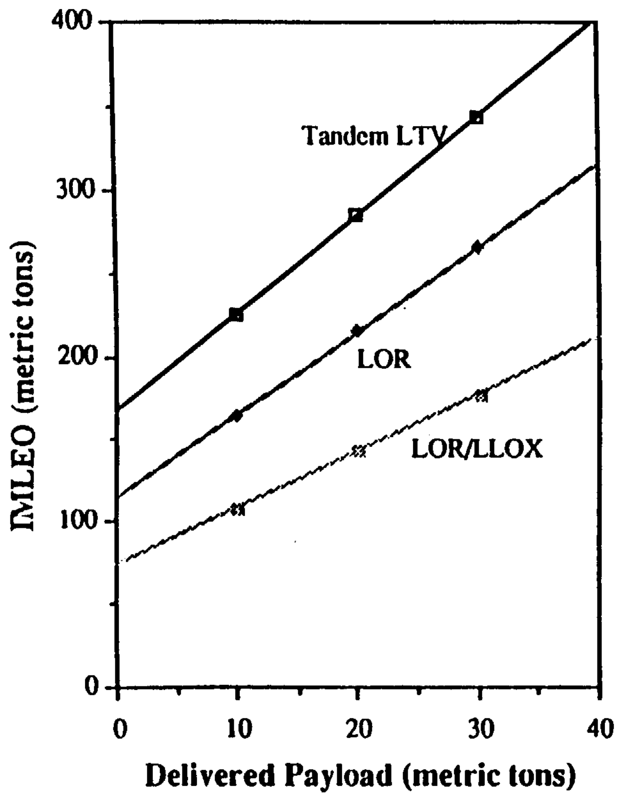
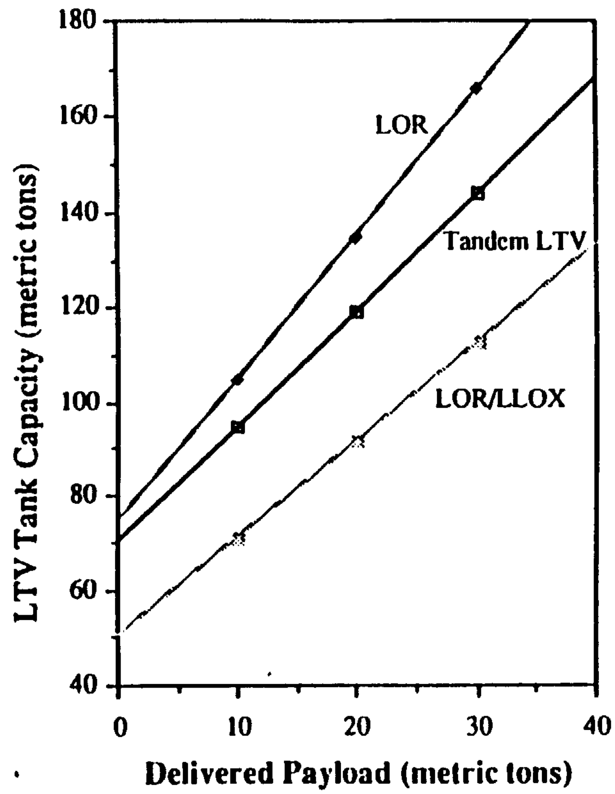
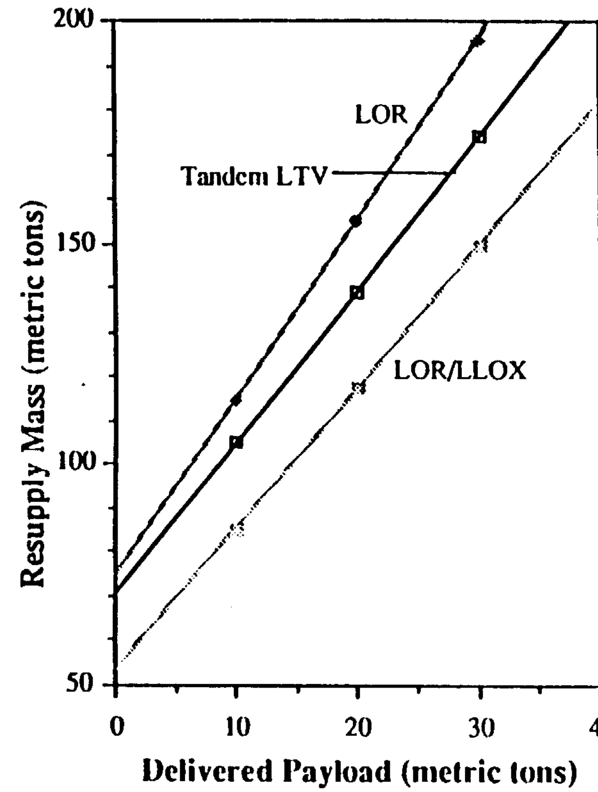
Lunar Modes Performance

Crew Mission, 1 Ton Returned







Notes:
- Crew size of 6
- LEV crew module of 4.847 tons
- 1 ton of additional consumables
- Isp of 475 seconds
- LTV crew module of 8.911 tons

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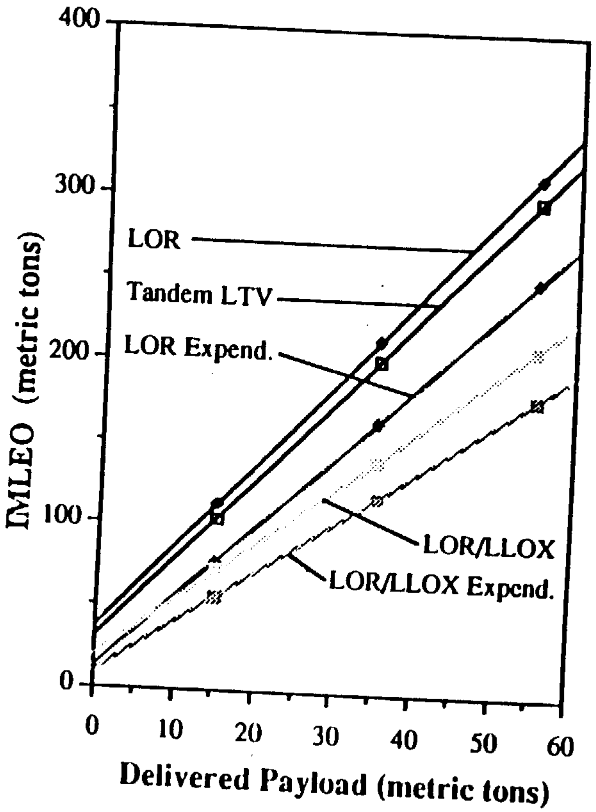
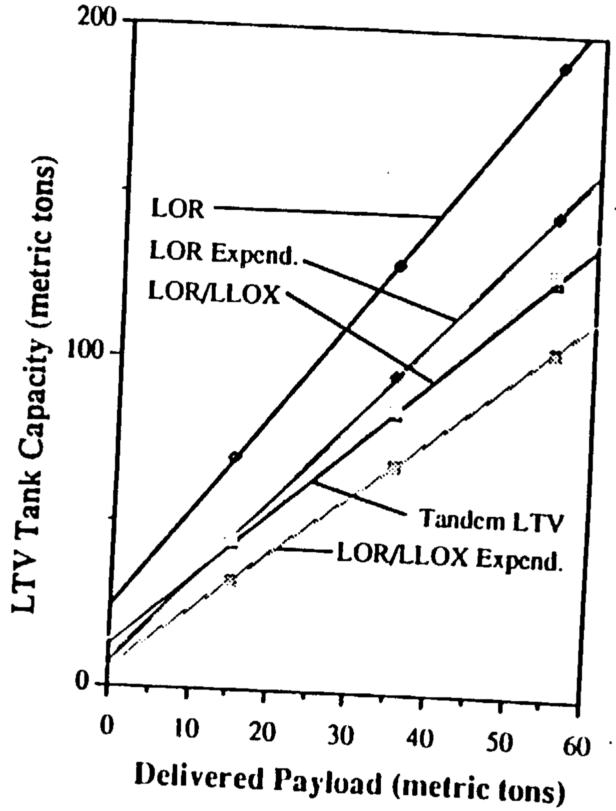
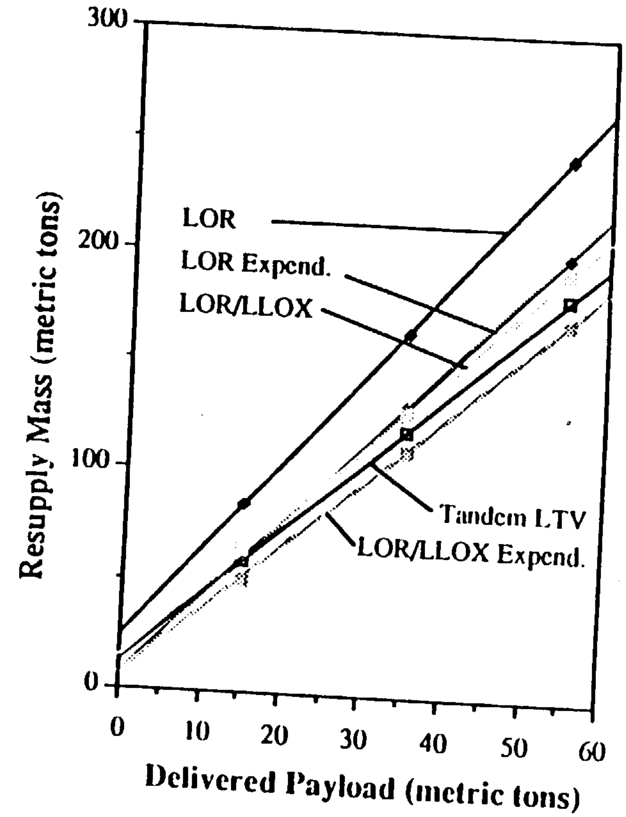
ADVANCED CIVIL
SPACE SYSTEMS

Lunar Modes Performance
Cargo Delivery Mission

BOEING







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